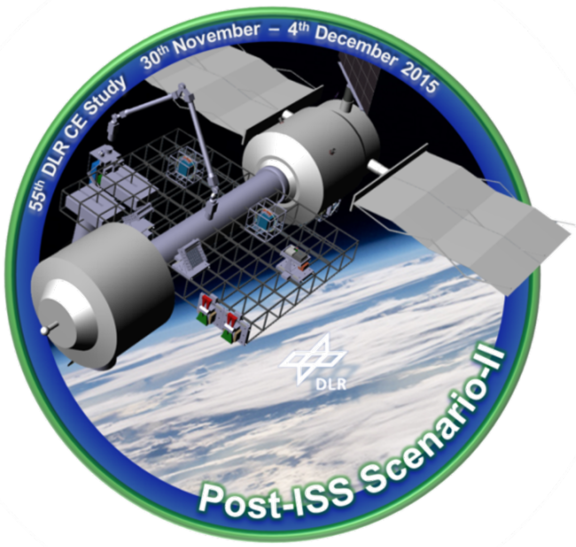
Institute of Space Systems
System Analysis Space Segment

Feasibility Study

Post-ISS Scenario-II

Concurrent Engineering Study Report



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Deutsches Zentrum für Luft- und Raumfahrt
German Aerospace Center

AIRBUS DEFENCE & SPACE

Bremen, 25th May 2018

Deutsches Zentrum für Luft und Raumfahrt e.V.
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1. Introduction

All Space Station partners agree to utilise the orbital research facility until 2020. NASA, Roscosmos and CSA announced to utilise the ISS even further until 2024. Whether this is politically, technologically and financially feasible for all partners is unknown. The overall question for all users is whether and how to continue with their investments. In general a transition to a new concept without critical know-how losses is around 10 to 15 years. Therefore, from a German/European point of view the technical layout, road mapping and development of a Human Spaceflight concept must be started now. The DLR project "Post-ISS" (system analysis study) can be understood as national preparatory work for establishing future programmes in the field of Human Spaceflight securing long-term research and astronautical activities in LEO. Corresponding questions focus on:

- How to continue with space research and space technology development after the ISS utilisation period (≥~2024)?

Therefore, the following objectives have been defined within the DLR study:

- Analysis of the ISS pros and cons (DLR internally) and recommendations based on Lessons-Learnt
- Market research of existing technologies / techniques
- Analysis of additional user demand and utilisation opportunities by including additional scientific disciplines and technological research
- Design of user conform infrastructure concepts to proceed with Human Spaceflight in LEO
- Analysis of re-usability of current architecture

The Concurrent Engineering (CE) study "Post-ISS Scenario-II" took place from 30th of November to 4th of December 2015 in the Concurrent Engineering Facility (CEF) at the DLR Bremen. The subsystem domains and disciplines were taken by Airbus Defence & Space, Consultants and mainly DLR staff. The goal of the study has been the investigation of the Free Flyer concept developed in the frame of the DLR-internal Post-ISS project.

1.1. General Background

For decades the International Space Station ISS demonstrates not only long-term international cooperation between 14 partner governments but also a significant engineering and programmatic achievement mostly as a compromise of budget, politics, administration and technological feasibility. Most ISS technologies are based on MIR and other previous experience. Due to high safety standards required for human space activities, latter are often conservative and new developments require patience and waiving 'state-of-the-art' technologies. A paradigm shift to more innovation and risk acceptance can be observed in the development of new markets by shifting responsibilities to private entities and broadening research disciplines, demanding faster access by users and including new launcher[1] and experiment facilitator companies[2] (see U.S.).

The research part of the systems-engineering study shows that space faring nations are developing their individual programmes for the time after ISS: NASA shifts LEO operations and utilisation to competing U.S. commercial companies while focussing on the next preparatory steps of Exploration (e.g. SLS, MPCV) of Asteroids, Moon and in long-term Mars. Russia plans new human rated space infrastructures at various optional locations (e.g. OKA-T Free Flyer) rather than committing to continue the utilisation of its dated ISS modules. In the field of human spaceflight China proceeds to go on with its Chinese Space Station (CSS) and prepares its next objective: the human Moon landing. Europe's human spaceflight partners seem to tend to the consideration of new platforms in LEO or cis-lunar space while utilising ISS as long as possible and necessary for the transition expected beyond 2024. Europe itself is interested in LEO and Human Spaceflight as discussed in ISECG, depending on the funding commitment. [RD-1]

In line with the space strategy of the German Government ISS follow-on activities should comprise of clear scientific objectives and technological key competences (e.g. robotic, internal and external structures, module/facility and experiment operations, interface systems (ATV)).

Therefore, DLR started to investigate future options by evaluating various LEO infrastructure concepts including opportunities for national realisation or international cooperation. A corresponding list of options can be found below. DLR scientists from various disciplines were asked to assess the usability of these options and design payloads based on their MIR and ISS experience and with respect to future scientific fundamental and technological research questions.

[1] US commercial launch providers currently are for example: SpaceX, Orbital Sciences.
2 European experiment facilitators AIRBUS and OHB tried the commercial approach but are still waiting for the success. US experiment facilitators are for example: Nanoracks, Kentucky Space and the mediator foundation CASIS. The only platform provider with a commercial approach is Bigelow.

1.2. Mission Outline

1.2.1. Mission Objectives

In the frame of the Post-ISS project, initiated by the DLR executive board, the following question shall be answered: _Assumed, Germany or Europe wants to continue the astronautic spaceflight in LEO: How could options look like, whilst considering the scientist's requirements?_

1.2.2. Study Goals

The requirements of the science community have been defined during the Post-ISS Payload CE-Study based on the User-Workshop (Cologne, May 2014). Now as done for the Base Station during the Post-ISS Scenario-I CE-study, these requirements shall be addressed in a more detailed architecture design regarding the Free Flyer. For that purpose the following shall be elaborated during the Post-ISS Scenario-II CE-study:

- Distribution of needed functions over modules (e.g. communication, ECLSS)
- Sizing of modules
- Layout of modules (primary structure and secondary structure, harness, accommodation, power, subsystems (including scientific payloads)) – shall Service Module contain tunnel?
- Integration of robotic / automation
- Find a formation with the Base Station that assures 1.) safety requirements and 2.) low fuel demand
- Operations scenario
- Design of infrastructure on ground
- Installation/ deployment scenario / launch
- Rough cost estimation

Thereby the following framework conditions shall be considered:

- Technical modular concept (separation of astronauts and experiments where required by science restrains; in failure case single modules' exchange is possible, optional autonomous operation of units (Habitat/ temporarily crewed Free Flyer)
- Political modular concept (countries/agencies can participate according to individual budget possibilities and science interest)
- Design (mainly) based on available technologies with participation of private partnerships
- User (science) requests for multiple disciplines (see details below)
- Reasonable costs for operations

1.3. Concurrent Engineering Approach

To investigate and define the technical concept of the Post-ISS Scenario-II a Concurrent Engineering (CE) Study at DLR Bremen has been conducted. The CE-study comprised the analysis and the development of all subsystems necessary for Post-ISS Scenario-II i.e. Systems, Thermal, Power, Crew Facilities, ECLSS, AOCS, Propulsion, Launch Scenario, Configuration, Structure, Debris & Radiation Protection, Robotic, Automation, Mechanisms, Mission Analysis, Communication, OBC, Ground Segment, Operations & Crew Support and Payload & Science.

The applied Concurrent Engineering (CE) process is based on the optimization of the conventional established design process characterized by centralized and sequential engineering (see Figure 1-1 top). Simultaneous presence of all relevant discipline's specialist within one location and the utilization of a common data handling tool enable efficient communication among the set of integrated subsystems (see Figure 1-1 bottom).

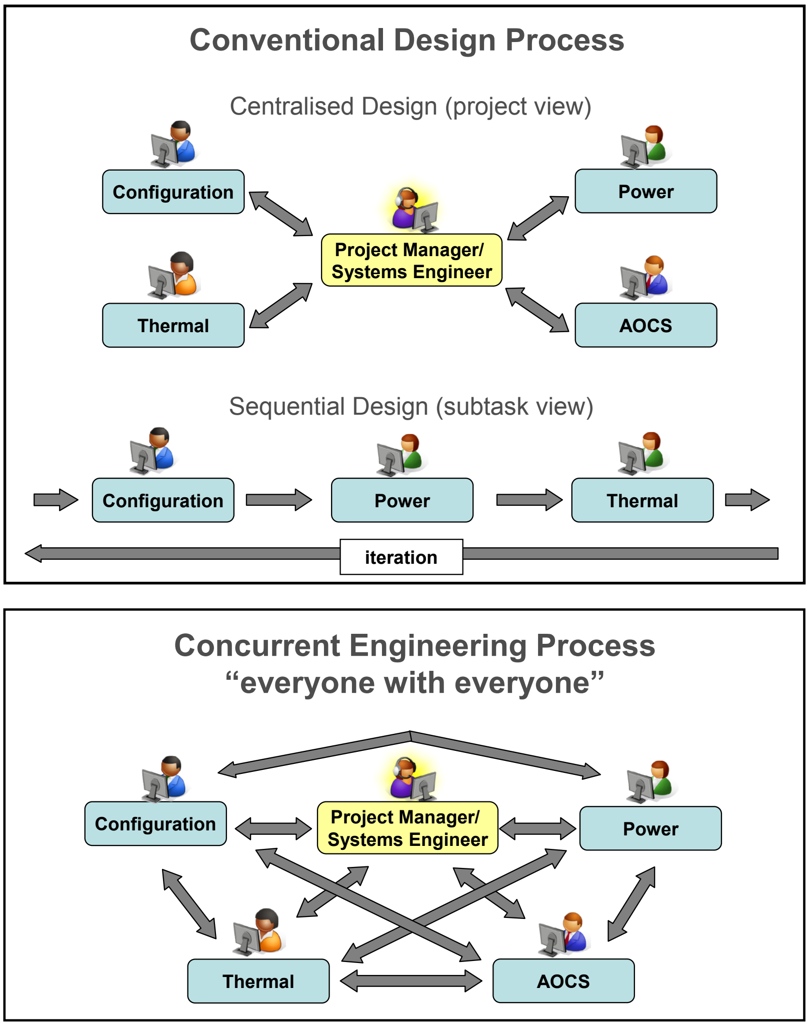


Figure 1-1: The Concurrent Design approach compared to projections of conventional design process.

The CE-Process is based on simultaneous design and has four phases ("IPSP-Approach"):

1. Initiation Phase (starts weeks/months before using the CE-facility):

- Customer (internal group, scientists, industry) contacts CE-team
- CE-team-customer negotiations: expected results definition, needed disciplines

2. Preparation Phase (starts weeks before using CE-facility):

- Definition of mission objectives (with customer)
- Definition of mission and system requirements (with customer)
- Identification and selection of options (max. 3)
- Initial mission analysis (if applicable, e. g. based on STK)
- Final definition and invitation of expert ensemble, agenda definition

3. Study Phase (1- 3 weeks at CE-Facility in site):

- K/O with presentations of study key elements (goals, requirements)
- Starting with first configuration approach and budgets estimates (mass, power, volume, modes, …) on subsystem level
- Iterations on subsystem and equipment level in several sessions (2- 4 hours each); trading of several options
- In between offline work: subsystem design in splinter groups
- Final Presentation of all disciplines / subsystems

4. Post Processing Phase:

- Collecting of Results (each S/S provides Input to book captain)
- Evaluation and documentation of results; transfer open issues to further project work

The DLR's Concurrent Engineering Facility in Bremen is derived from the Concurrent Design Facility at ESA's ESTEC (European Space Research and Technology Centre), which has already been in operation for more than ten years. The CEF has one main working room where the whole design team can be assembled and each discipline is supplied with an own working with special design tools and a common design and data model. Three screens allow display of data in front of the team. Further working positions are provided in the centre of the working area and are usually reserved for customers, advisors, guests as well as the team leader. Two more splinter rooms provide the design team with separated working spaces where sub-groups can meet, discuss and interact in a more concentrated way.

Figure 1-2: Concurrent Engineering Facility during CE-study phase at DLR Bremen

The major advantages of the Concurrent Engineering (CE)-process are:

- Very efficiency regarding time, cost & results of a design activity
- Assembly of the whole design team in one room facilitates direct communication and short data transfer times, supported by a moderator
- The team members can easily track the design progress, which also increases the project identification
- Ideas and issues can be discussed in groups, which brings in new viewpoints and solutions; incl. avoidance and identification of failures and mistakes

1.4. Document Information

This document summarizes the progress and results of the DLR Concurrent Engineering study about the Post-ISS Scenario-II, which took place from 30^{th} of November to 4^{th} of December 2015 in the Concurrent Engineering Facility of the DLR Institute of Space Systems in Bremen. The single subsystems or domains as investigated during the study are covered in individual chapters, which explain the study progress, elaborate on decisions and trade-offs made during the study and also design optimizations. If not allowed by the DLR directorate, the document should not be distributed outside DLR before November 2017 (study team members excluded). A comprehensive documentation of the overall Post-ISS project in German can be found under DLR-RY-Post-ISS Projektbericht: AP1000 "ISS-Analyse und Lessons Learnt"; DLR-RY-Post-ISS Projektbericht: AP2000 "Konzept-bewertung", DLR-RY-Post-ISS Projektbericht: AP3000 "Mögliche Anwendungen & Nutzlasten", DLR-RY-Post-ISS Projektbericht: AP4000 "Szenarienentwurf".

2. Systems

2.1. System Architecture

During the CE-study Post-ISS Scenario-I the Orbital-Hub's Base Station (Figure 2-1, left) has been laid out consisting of a docking node, a service module and an expandable habitat. Focus of the Post-ISS Scenario-II is on the Free-Flyer (Figure 2-1, right) consisting of a pressurised laboratory (PL), an external science platform (ESP) and a service module (SE).

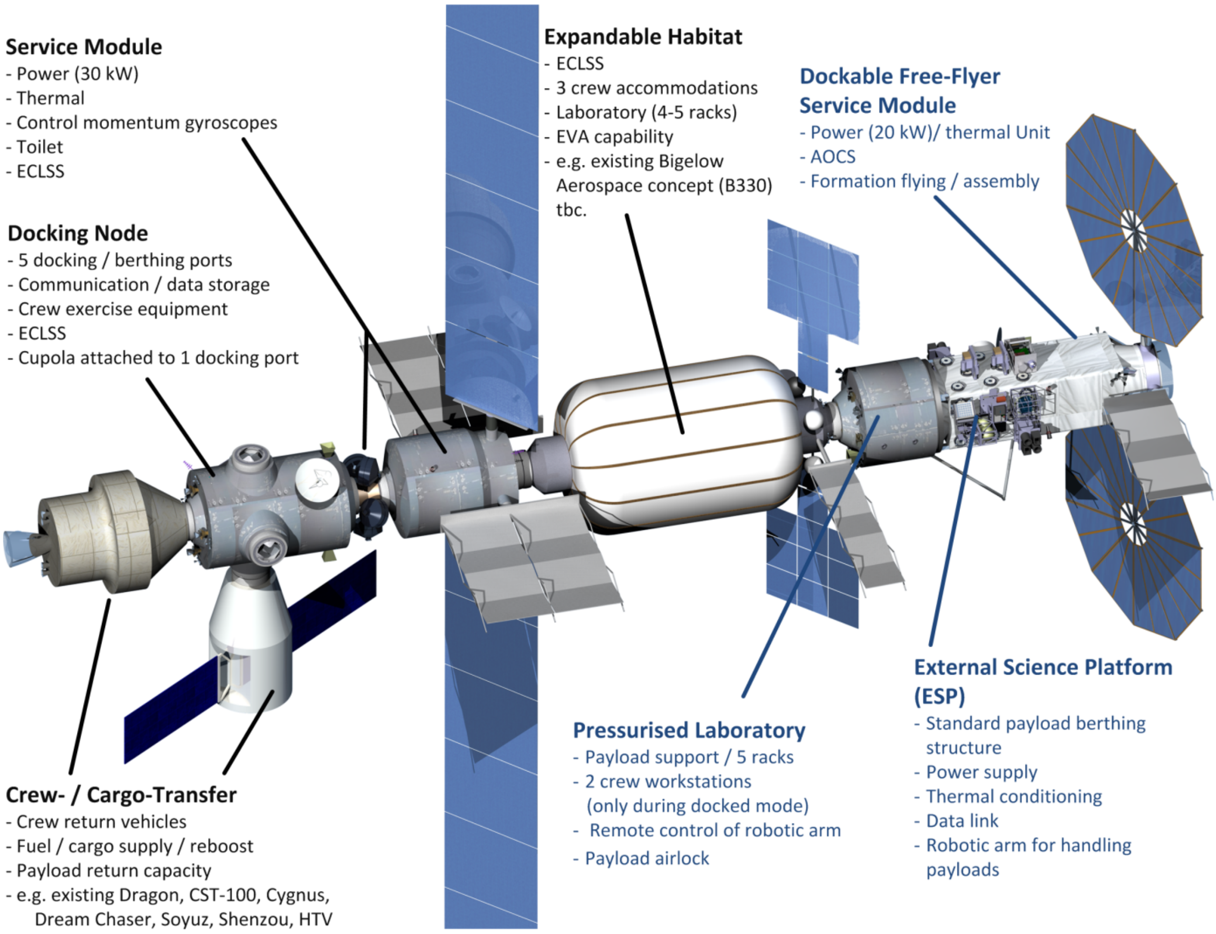


Figure 2-1: DLR Orbital-Hub architecture (left/black: Base Station; right/blue: Free Flyer).

2.2. Mission Requirements

In preparation of the CE-study the following mission requirements have been defined:

Table 2-1: Post-ISS Scenario-II – Mission Requirements.

No.	Requirements
MI-010	The Free Flyer shall maintain an orbit altitude of 400 km +/- 50 km
MI-020	The Free Flyer shall maintain an orbit inclination of about 51.6 deg
MI-030	The Free Flyer shall be flexible in orientation (e.g. oriented nadir as baseline, inertial (e.g. star pointing) for a period of tbd days, orientation for minimum disturbance and best µg-condition (in order of $<10^{-6}$ g) for a period of 14 days).
MI-040	The Free Flyer shall be able to act as active part for the assembly of the Base Station.
MI-050	The Free Flyer shall dock to the Base Station every 3 month **(average)** for a period of two weeks (docking is not intended during crew exchange).
MI-060	**The last phase of rendezvous has to be completed within 6 hours (one shift of operations)**

Hereby the ISS orbit (see Figure 2-2) was chosen to be able of using the Free Flyer already for ISS or remaining modules and to benefit from the well-established and proven ground network and launch facilities. Furthermore this orbit also was of interest for the Earth observation community as a complement to higher inclined and geostationary missions.

Information in bold letters was added during the CE-study.

The requirement to fulfil the rendezvous within 6 hours (MI-060) was developed by the help of the operations domain and was one driver of the propulsion system design.

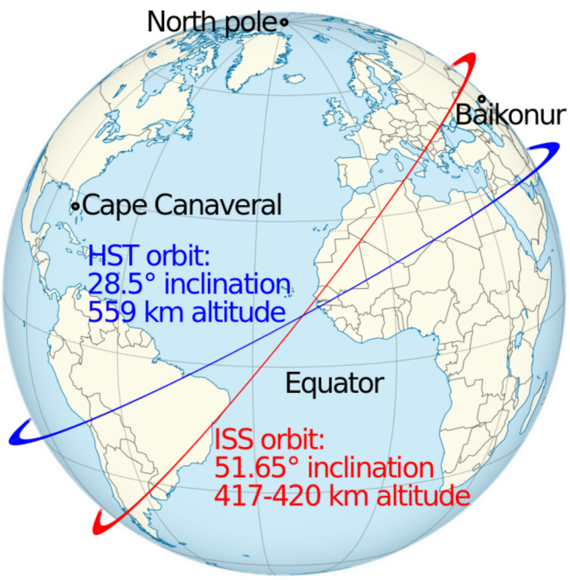


Figure 2-2: ISS orbit representation [TUBS, Luan; CC-BY-SA-3.0]

2.3. System Requirements

In order to dimension the spacecraft's subsystems and to fulfil the science constraints the following system requirements has been defined and fulfilled by the design team:

Table 2-2: Post-ISS Scenario-II – System Requirement.

No.	Requirements
ST-010	The design shall be based on technologies that are available 2025
ST-020	The system's lifetime shall be 15 years.
ST-030	The Free Flyer shall have a diameter <5 m, a length <14 m and a **launch** total mass <19,000 kg; **total on-orbit mass is to be <25,000 kg**
ST-040	The Free Flyer shall consist of a pressurized laboratory, an external science platform and a service module (for power/thermal, AOCS and manoeuvres)
ST-050	The international docking standard shall be used (IBDM: ∅80 cm)
ST-060	As part of Option A.4 the Free Flyer shall be able to dock with the Base Station and will be provided with ECLSS for the pressurized part of the Free Flyer by the Base Station
ST-070	The station shall be laid out for a crew of temporarily up to two persons in pressurized part of docked Free Flyer
ST-080	The pressurized part shall be able to house 12 ISPRs, from which some space will be needed for optical instruments and for a payload airlock.
~~ST-090~~	~~The pressurized part shall contain three fused quartz windows (60 mm) for optical instruments (nadir oriented).~~
ST-100	The external science platform shall be able to accommodate **area of** 10 JEM EF **equivalent** payloads (8 nadir; 2 pointing to free space) and an area of 8 m² for smaller payloads (<500 mm height)
ST-101	The JEM EF payloads shall be provided with a cooling liquid and power (both in total 5 kW), Ethernet (≥300 Mbit/s each), Video and 1553 data connectors.
ST-102	The area for smaller payloads (<500 mm height) shall provide power (in total 3 kW), Ethernet (≥300 Mbit/s each), Video and 1553 data connectors for 8 payloads and in addition cooling (max. 3 kW) for at least two payloads.
ST-103	**The ESP shall allow mounted P/L launch loads, which are not exceeding maximum occurring on-orbit loads**

ST-110	Astronautical maintenance can be set for **Pressurised Lab, no other areas are accessible or maintainable from interior; external access is only contingency**
ST-120	**The Free Flyer shall provide 20 kW of average power for housekeeping and payload (not including charging pwr)**
ST-121	**The Free Flyer shall be able to survive a loss of power generation for 1.5 orbits (system powered in survival mode)**
ST-130	**The Free Flyer is designed one-failure tolerant for normal equipment and two-failure tolerant for human spaceflight related equipment**
ST-140	**The Free Flyer shall allow a coolant loop temperature between 10-30°C**

Information in bold letters was added during the CE-study.

For the calculation of moments of inertia and fuel consumption or tank capacity an agreed maximum in orbit mass was required (see ST-030).

ST-090 was not fulfilled due to safety and cost concerns.

ST-103 answers the question whether payload can already be installed on the ESP before launch.

There was a discussion if the SE might be also maintainable from inside by astronauts via a tunnel connection to the PL. It was discarded due to increasing complexity (see ST-110). As a consequence to that, critical equipment such as transponders were positioned in the PL instead of the SE and the Control Momentum Gyros, which failed several times on the ISS, were placed in such a way to the SE that they could be reached by the robotic arm of the ESP.

In contrast to the ISS minimum cooling loop temperature of -20°C during the study it was aimed for higher temperatures in order to keep the Free Flyer's thermal system and radiator compact (see ST-140).

2.4. Baseline Design

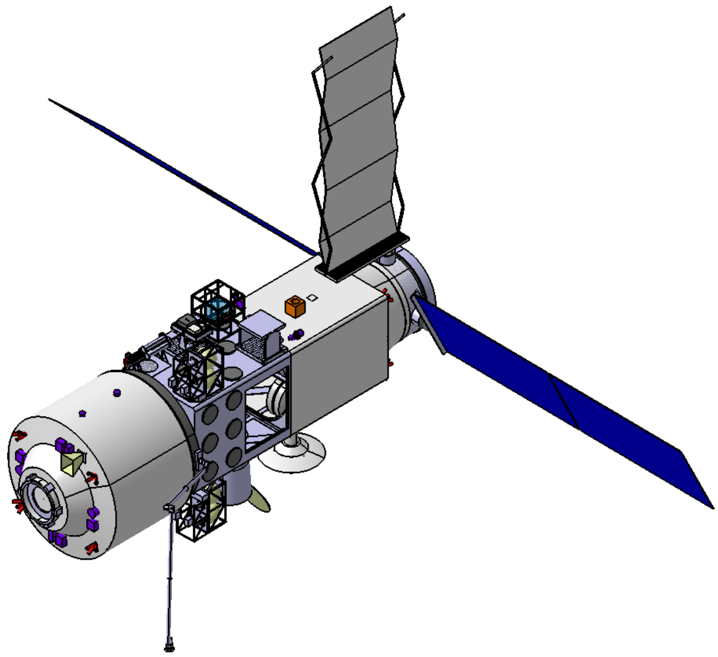


Figure 2-3: Baseline concept of the Free Flyer.

The Free Flyer is part of the Orbital Hub concept (see Figure 2-1) in response to the scientific user requirements. It is intended to fly uncrewed in a safe formation to the Base Station for e.g. three months periods until it can be maintained or reconfigured when docked to the base for short duration. In analogy to the Base Platform, it also requires a service module for attitude and orbit control and also for formation flying and independent power and thermal control. Furthermore, it contains a pressurised module for µg-research which can be accessed when docked to the Base Station (e.g. via the Docking Node or via the Expandable Habitat module) or to a crew vehicle. The external platform is the centre of the Free Flyer (see Figure 2-3). It has a standardized berthing structure for external payloads and provides power, data and thermal conditioning. The Free Flyer will most likely fly with the instruments pointed nadir, but in principle, is free to change attitude for certain periods depending on user requirements. The Free Flyer's External Platform is designed as a rigid rectangular truss structure covered with multi-layer insulation. The main volume of the payload airlock is located inside this structure and can be reached through a cut-out by the robotic arm. This manipulator is moving along a rail around the structure to place different payloads onto the four sides of the platform with respect to their desired viewing direction. As the Free Flyer's Service Module does not need to be pressurised, it has been redesigned using the same truss approach as the External Platform and by this facilitating the mechanical design for stiffness and launch load transfer through the overall structure. Robotic arm interfaces are foreseen to handle the payloads on the platform, which is based on the Post-ISS Payload CE-study. Furthermore, the Free Flyer is intended to support the assembly of the Base Station by being the active part of automated docking, since there is currently no similar vehicle like the U.S. Space Shuttle available. The overall dimensions of the Free Flyer in stowed configuration (retracted photovoltaics and radiator wings) have been optimised to be in line with the launch scenario using a single Ariane 64.

2.4.1. Power Modes

During the study the following two operational modes have been defined in order to lay out the power subsystem. It is assumed that the Base Station could provide additional power if necessary when being docked and provide the ECLSS function for the PL during crewed activity.

Table 2-3: Modes of Operation for the power layout of Post-ISS Scenario-II.

Mode Name	Abbreviation	Description	Reference Duration
Standard Mode	StM	core operations including science; no crew on-board, drag compensation, nadir-velocity fixed or inertial	2 weeks
Survival Mode	SurvM	no power is generated, power consumption is minimized	2 days

The overall power budget can be found in the power chapter in Table 11-1.

2.4.2. Mass Distribution

In the following the overall mass budget of the final Free Flyer design is presented.

Since the launch mass was restricted for a single launch to 19 t (req. ST-030), there is a mass budget given for the launch configuration in Table 2-4 with an overall launch mass of 18.7 t. For that a fictive mass of 813.6 kg including a systems margin of 20% is foreseen as already installed payload on each, the PL and the ESP. The contribution of each subsystem to the overall mass is depicted on the left pie chart and the distribution over PL, ESP and SE on the right pie chart of Figure 2-4.

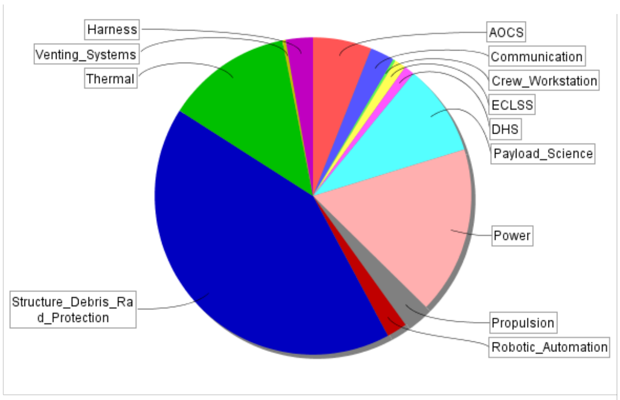
Due to mass limitations but also stiffness requirements (req. ST-103) during launch, most of the payloads have to be installed on orbit launched via separate cargo flights. For the maximum equipped on orbit configuration the mass budget is given in Table 2-5 accommodating over 8.3 t of strawman payload. When the overall system margin of 20% is not taken into account for the payloads, one can subtract 1668 kg from the given overall mass of 27082.49 kg. The resulting overall on orbit mass of the fully equipped Free Flyer with 25.4 t comes close to the postulated maximum of 25 t (req. ST-030). The resulting contribution of each subsystem to the overall mass is depicted on the left pie chart and the distribution over PL, ESP and SE on the right pie chart of Figure 2-4.

Table 2-4: Mass budget of Post-ISS Scenario-II in launch configuration.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
AOCS	733.51	19.13	140.30	873.81	6.00
Communication	311.00	15.87	49.35	360.35	2.47
Crew_Workstation	30.00	20.00	6.00	36.00	0.25
DHS	141.00	8.40	11.85	152.85	1.05
ECLSS	156.40	9.49	14.84	171.24	1.18
Harness	367.48	10.00	36.75	404.23	2.78
Payload_Science	1130.00	20.00	226.00	1356.00	9.31
Power	2271.14	9.61	218.36	2489.50	17.09
Propulsion	352.23	12.51	44.06	396.28	2.72
Robotic_Automation	267.00	13.75	36.70	303.70	2.09
Structure_Debris_Rad_Protection	5110.00	19.37	990.00	6100.00	41.88
Thermal	1666.00	12.40	206.60	1872.60	12.86
Venting_Systems	40.80	16.03	6.54	47.34	0.33

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	12576.56			14563.91	
System margin:		20.00		2912.78	
Total dry mass with system margin:				**17476.69**	
Propellant:				1100.00	
Adapter mass:				125.00	
Launch Mass:				**18701.69**	
Max launcher capacity:				19000.00	
Buffer to launch mass:				298.31	

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
ESP_External_Science_Platform	2372.16	17.56	416.49	2788.65	19.15
PL_Pressurized_Lab	4744.74	17.34	822.53	5567.27	38.23
SE_Service_Module	5459.66	13.71	748.33	6207.99	42.63



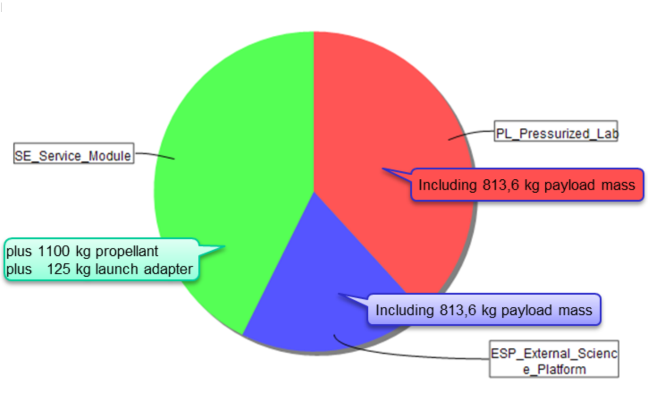


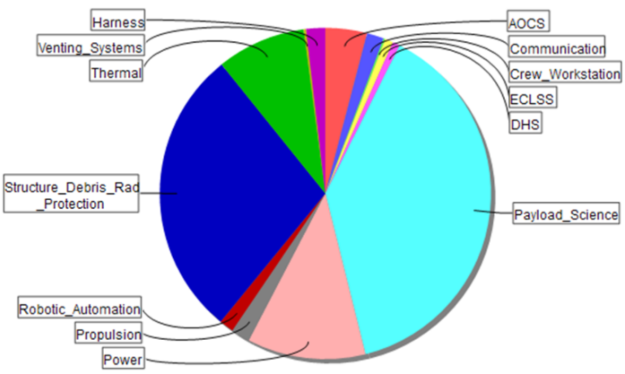
Figure 2-4: Mass distribution of the Free Flyer in launch configuration (left: subsystems; right: modules).

Table 2-5: Mass budget of Post-ISS Scenario-II including 10 t of payload.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
AOCS	733.51	19.13	140.30	873.81	4.06
Communication	311.00	15.87	49.35	360.35	1.67
Crew_Workstation	30.00	20.00	6.00	36.00	0.17
DHS	141.00	8.40	11.85	152.85	0.71
ECLSS	156.40	9.49	14.84	171.24	0.79
Harness	367.48	10.00	36.75	404.23	1.88
Payload_Science	6950.00	20.00	1390.00	8340.00	38.70
Power	2271.14	9.61	218.36	2489.50	11.55
Propulsion	352.23	12.51	44.06	396.28	1.84
Robotic_Automation	267.00	13.75	36.70	303.70	1.41
Structure_Debris_Rad_Protection	5110.00	19.37	990.00	6100.00	28.31
Thermal	1666.00	12.40	206.60	1872.60	8.69
Venting_Systems	40.80	16.03	6.54	47.34	0.22

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	18396.56			21547.91	
System margin:		20.00		4309.58	
Total dry mass with system margin:				**25857.49**	
Propellant:				1100.00	
Adapter mass:				125.00	
Launch Mass:				**27082.49**	
Max launcher capacity:				19000.00	
Buffer to launch mass:				-8082.49	

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
ESP_External_Science_Platform	6757.16	19.14	1293.49	8050.65	37.36
PL_Pressurized_Lab	6179.74	17.95	1109.53	7289.27	33.83
SM_Service_Module	5459.66	13.71	748.33	6207.99	28.81



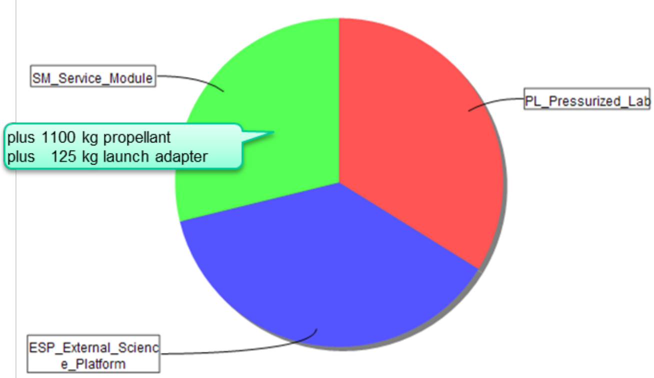


Figure 2-5: Mass distribution of the Free Flyer including 10 t of payload (left: subsystems; right: modules).

2.5. Harness

- At this early stage of the system definition the harness mass can only be defined relative to the system complexity and dimension. The harness mass is mainly driven by
  - Harness length (dimension of spacecraft)
  - Harness diameter dependent on interface voltage level for power harness
  - Signal harness complexity
  - Redundancy requirement
- Based on an assessment for existing man-rated spacecraft a typical ratio between the avionics equipment net mass and the harness mass could be identified which is approx. 5%

With a more mature configuration the harness mass has to go through other iterations.

Table 2-6: Overall Harness mass.

Name	Current Value	Unit
mass_total	367.48	kg
mass_total_wMargin	404.23	kg
power_avg_wMargin	0.00	W
▷ power_energy_wMargin	0.00	J

2.6. To be studied / additional Consideration

- Check if Payload interfaces/ positioning obstructing robotic arm movement on ESP
- Investigate alternative to ammonia for TCS due to health and safety reason
- Check saving potential of propellant consumption per docking manoeuvre

2.7. Summary

During the study week a detailed architecture design of the Free Flyer has been created, considering the science requirements from the Post-ISS Payload CE-study and the Base Station CE-study (Post-ISS Scenario-I). Functions, e.g. communication or ECLSS, have been allocated to the Free Flyer’s modules PL, ESP and SE. The modules have been laid out and sized regarding primary and secondary structure, harness, accommodation, power and further subsystems including scientific strawman payloads. Thereby an optic window in the PL or a tunnel towards the SE has been rejected, whilst automation and a flexible external robotic arm have been integrated into the design. Also a formation with the Base Station has been defined, assuring the safety requirements and moderate fuel demand.

3. Configuration

3.1. Requirements and Design Drivers

The main requirements for the configuration design of the POST-ISS Free Flyer (FF) are:

- The FF shall consist of the following three parts: Pressurized Laboratory (PL), External Science Platform (ESP) and Service Module (SE).
- The FF shall accommodate the demanded components of all subsystems.
- Desired / required positions and line of visions shall be respected for all components with special demands on their configuration.
- The FFs overall dimensions in launch configuration shall be in line with the Single Launch scenario assuming an ARIANE 6 fairing (approx. Ø5 m x 19.8 m).
- The PL shall use an IBDM adapter to connect with the Habitat Module.
- The ESP shall be equipped using robotic manipulation without the need of additional EVAs.
- The SE shall provide sufficient areas for photovoltaics in orbit configuration (including sun tracking in alpha and beta angle).
- The SE shall provide sufficient areas for radiators in orbit configuration.

The main design drivers for the FF configuration design are:

- The available volume for the launch configuration inside the assumed ARIANE 6 fairing
- The required photovoltaics / radiator areas in orbit configuration
- The decision on pressurized or unpressurized and by that on the maintainability of the SE
- The design of the ESP and trade between deployable or rigid structure type
- The strategy for the equipment of the ESP and by that the accommodation of the required airlock inside the PL
- The selection of a mobility unit for the robotic arm to reach all areas of the ESP
- Structural design considerations concerning load transfer

3.2. Baseline Design

Starting from an existing design from former studies, the FF configuration evolved during the study especially for the ESP and the SE. The following chapters describe the overall design with a focus on the newly designed FF part and a detailed description of the three modules of the FF.

3.2.1. Overall System Configuration

The FF is part of the Orbital-Hub concept for POST ISS scenario. It is the active component of the station and thereby is able to perform manoeuvres and dock to different Nodes. The following Figure 3-1 and Figure 3-2 show the Orbital-Hub in different configurations with the FF docked to the Habitat module or to the Docking Node.

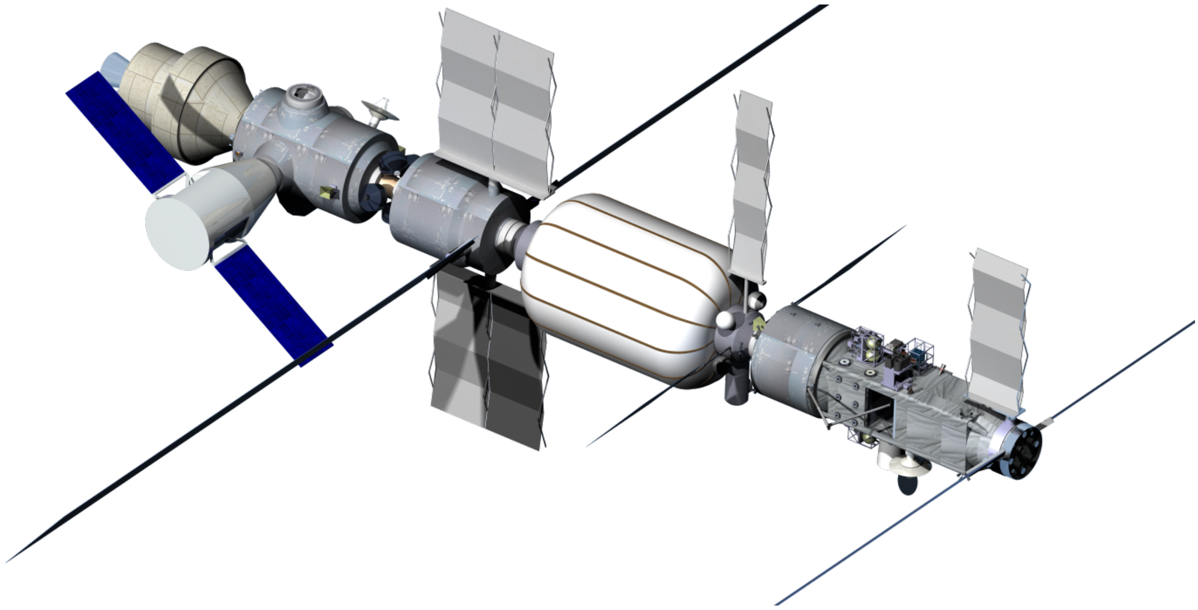


Figure 3-1: Complete Orbital-Hub assembly with Free Flyer docked to Habitat Module

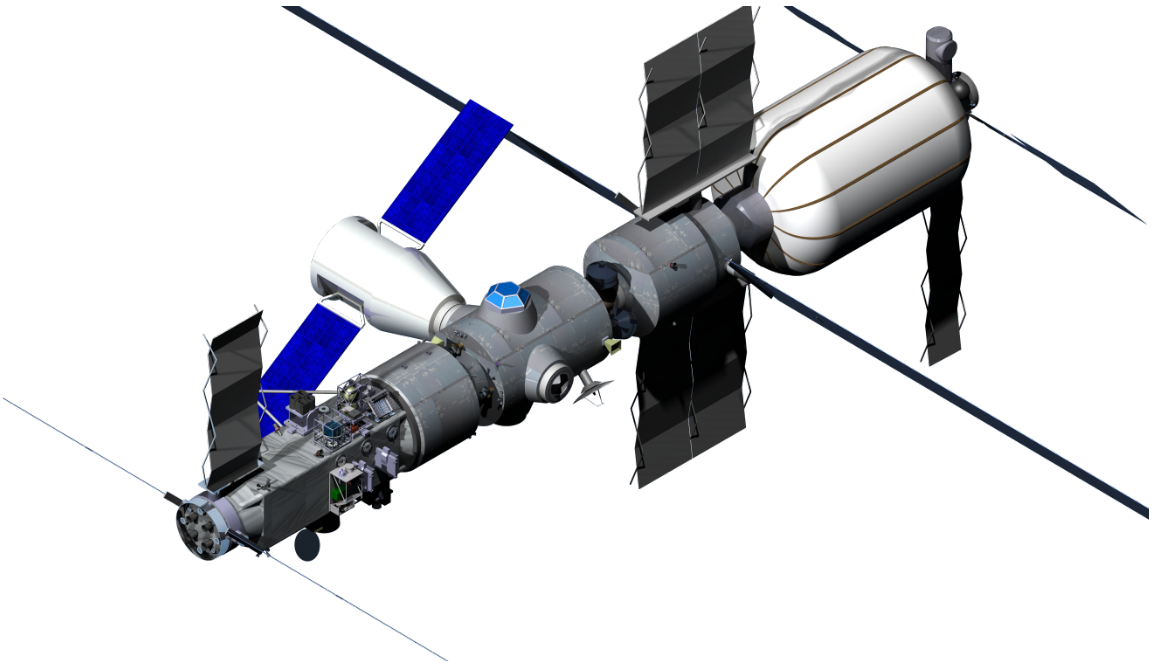


Figure 3-2: Complete Orbital-Hub assembly with Free Flyer docked to Docking Node

As a comparison of sizes between the Station and the FF modules, Figure 3-3 visualizes the main dimensions of the complete configuration. With a maximum span-width of 71 m and an overall length of 53 m (including rescue capsule), the Orbital-Hub has more or less the size of half a field of a typical soccer ground. By introducing more sophisticated folding mechanisms for the solar wings, at least the span-width is likely to be reducible.

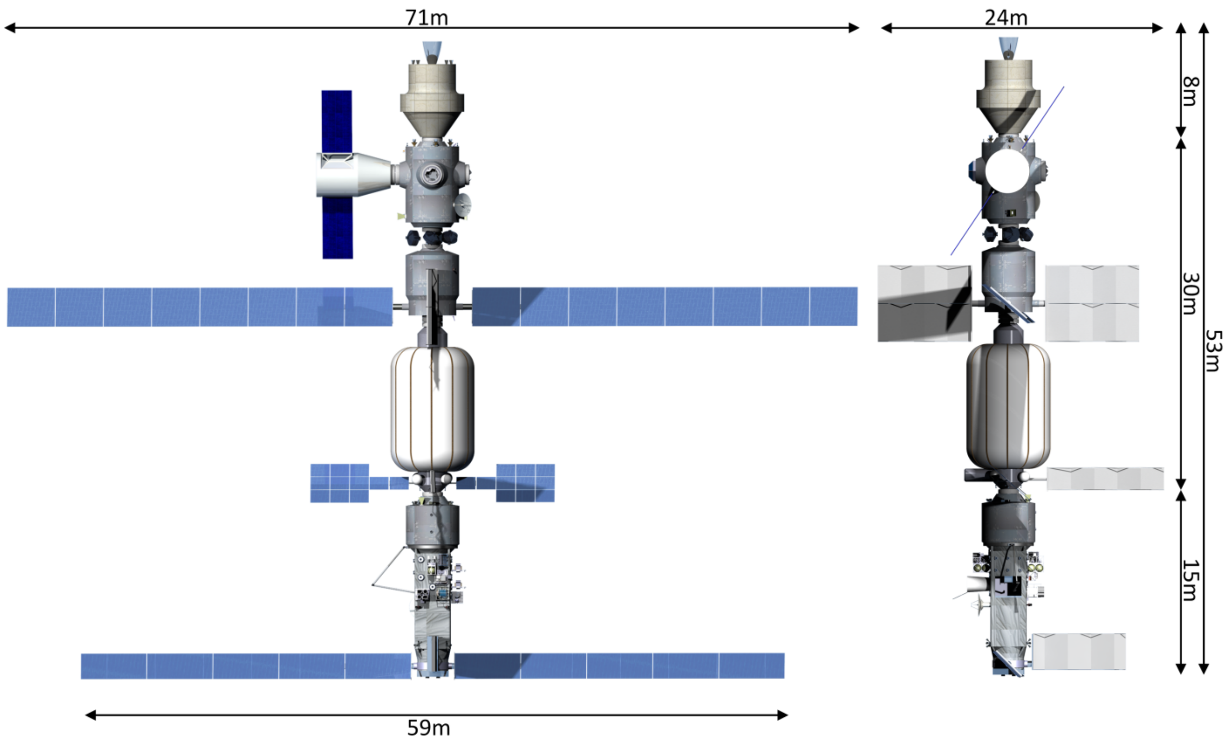


Figure 3-3: Main dimensions of complete Orbital-Hub assembly with Free Flyer docked to Habitat Module

3.2.2. Free-Flyer

Due to the single-launch scenario and therefore the limited available volume for the spacecraft inside the launcher's fairing, the FF has to be designed with deployable solar and radiator wings. Moreover, the ESP is assumed to be mostly unequipped during launch and the robotic arm has to be in folded configuration. Once in orbit, the wings and robot arm are released and fold out to operational flight configuration (c.f. Figure 3-4, Figure 3-5 and Figure 3-6). Afterwards, the ESP can be equipped via robotic manipulation. For a detailed description of each of the FF's three main modules, please refer to the following chapters.

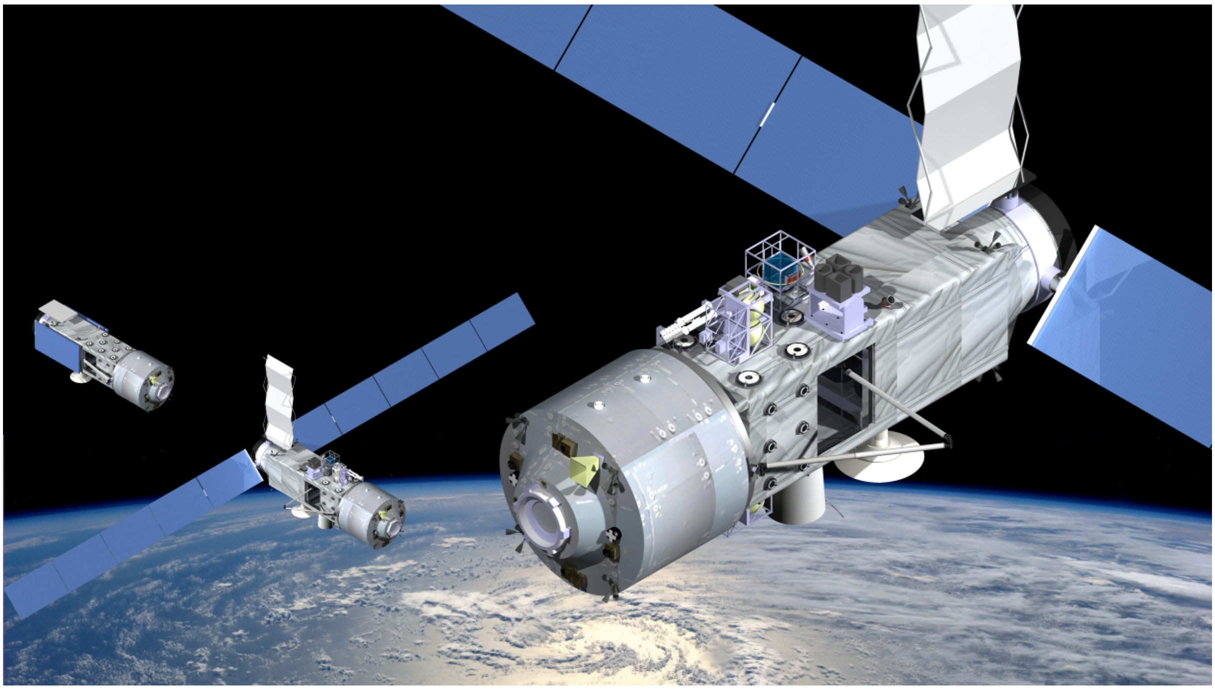


Figure 3-4: Configuration of Post-ISS Scenario-II

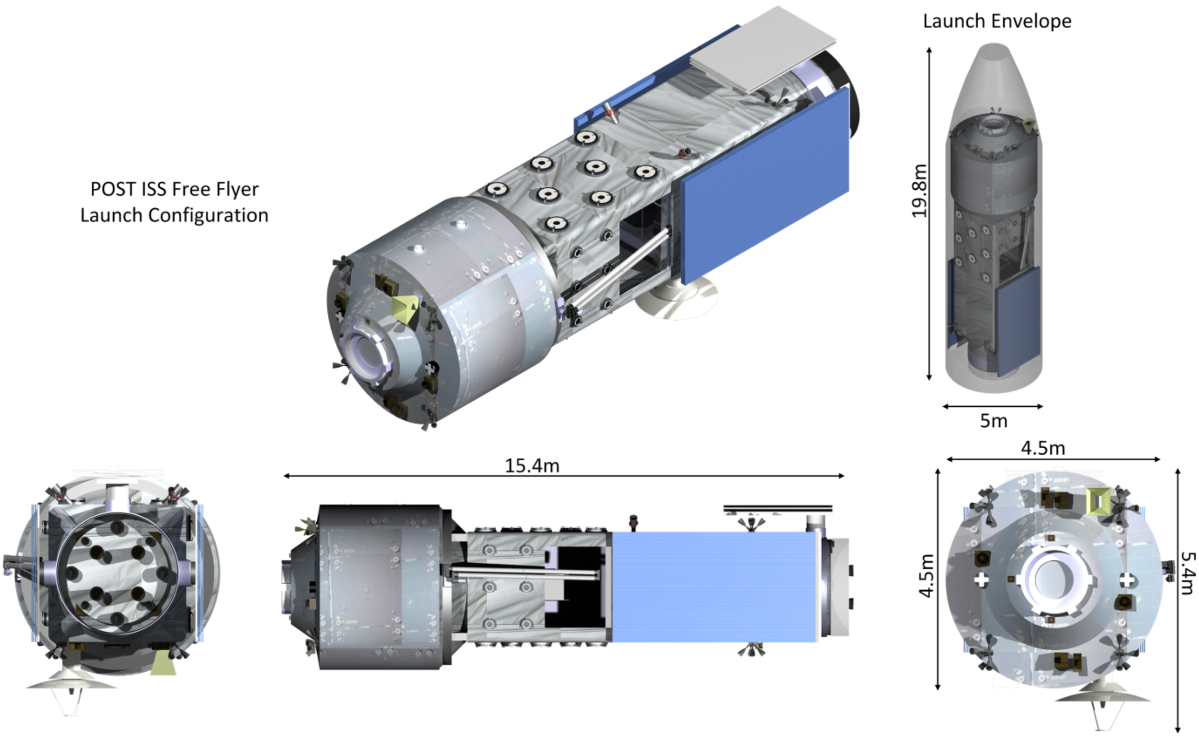


Figure 3-5: Complete Free Flyer views and outer dimensions in launch configuration

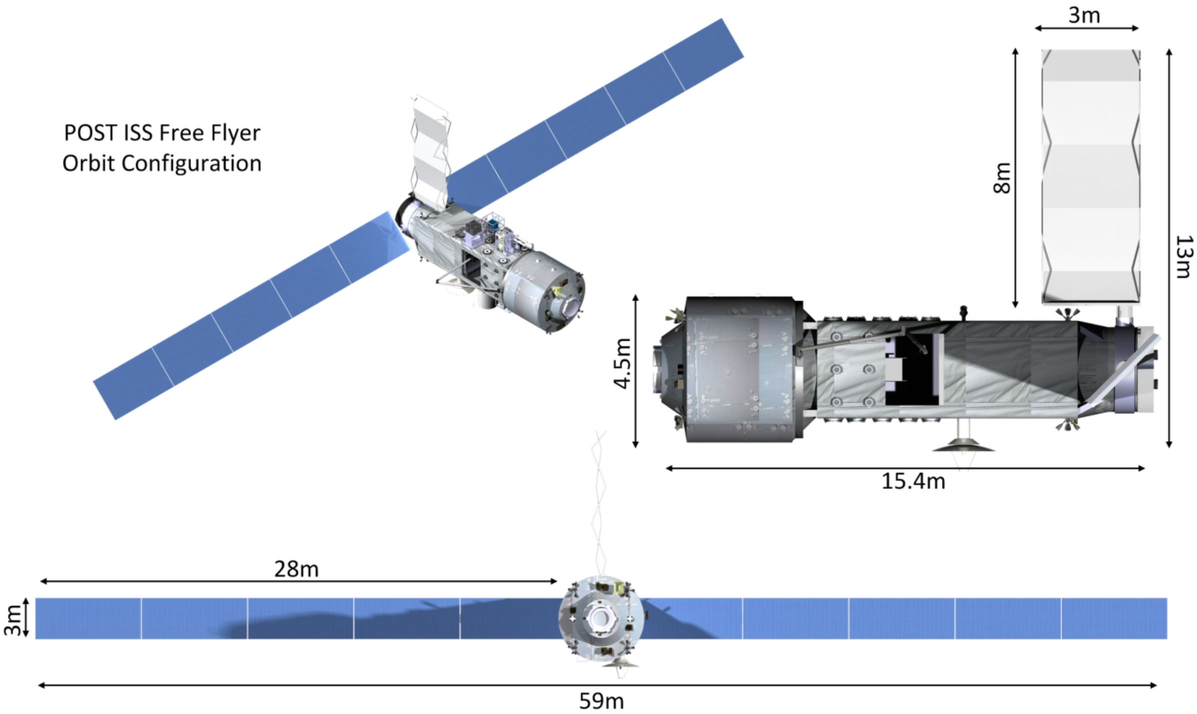


Figure 3-6: Complete Free Flyer views and outer dimensions in orbit configuration

3.2.3. Pressurized Lab

The outer shape of the PL has basically been untouched compared to the former studies. A typical barrel/cylindrical shaped structure as known from existing ISS modules (e.g. Columbus) has been selected and sized to meet the requirement of the accommodation of 5 payload ISPRs plus another 7 ISPRs for e.g. crew equipment and avionic/bus components.

One major decision concerning this module was to shift the needed Airlock for equipping of the ESP to the outside of the PL (and by that to the inside of the ESP structure). Thereby, the available volume could exclusively be used for the needed ISPRs. The outer dimensions can be found in the following picture Figure 3-7. The Airlock is 2m long and 1.7m in diameter and thereby in line with the existing design of the Japanese KIBO airlock (c.f. [RD 5]). These dimensions are assumed to be suitable for the transport of payload with the envelope of required JEM EFU container types.

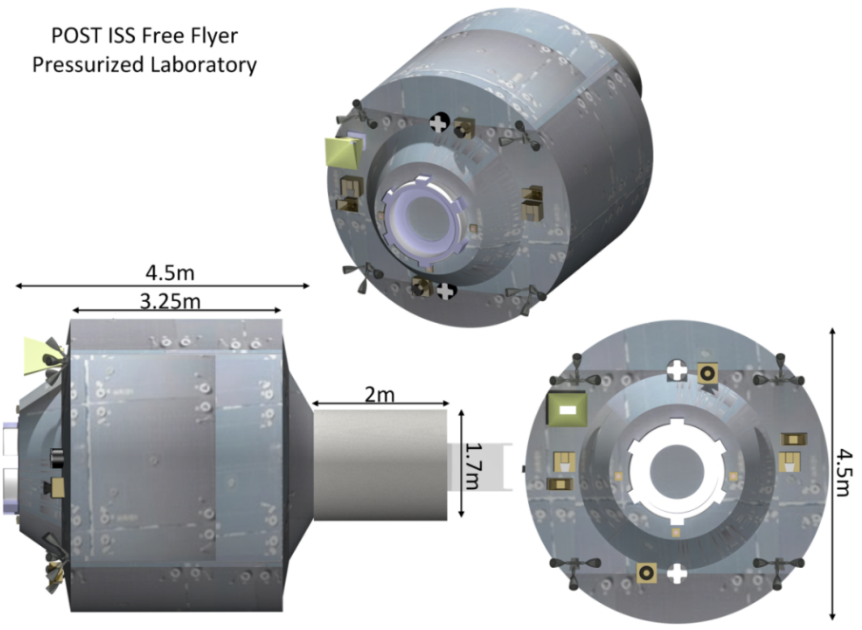


Figure 3-7: Free Flyer Pressurized Lab (including airlock) views and outer dimensions

The main components on the outside and inside of the module can be seen in Figure 3-8. Most of the outer components are located at the front and are used mainly for communication and attitude determination plus control during rendezvous and docking with the habitat part. The 12 ISPRs are distributed as following: 5 required MUMS payload racks, 1 crew equipment rack (which is supposed to house all panels for e.g. lighting and environmental control, a toolbox and which can also be equipped with the work station desk) and another rack for avionic and bus components, recently mainly occupied by AOCS and DHS. The remaining 5 racks are still free for further equipment, stowage or experiments.

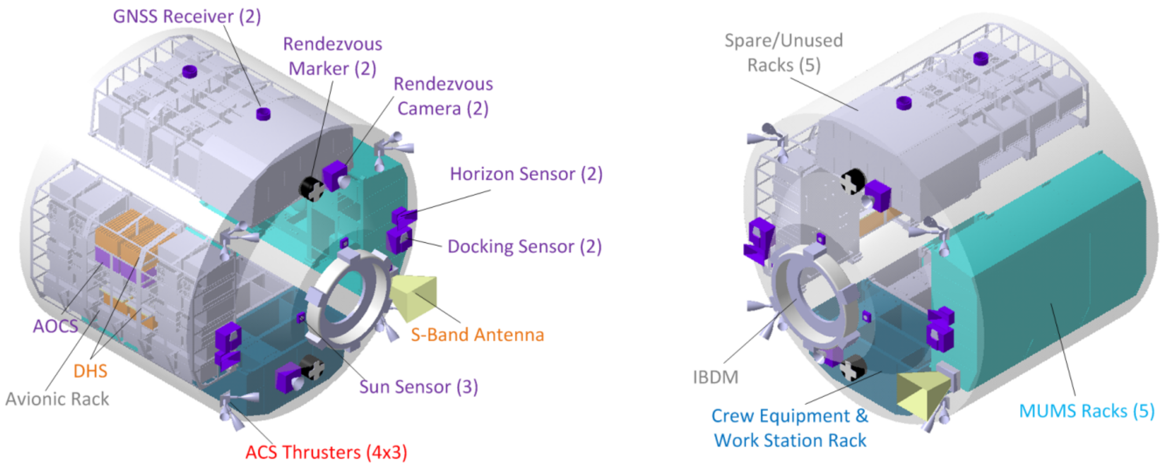


Figure 3-8: Free Flyer Pressurized Lab interior view including main components

3.2.4. External Science Platform

The ESP has been heavily revised compared to the initial configuration. The former concept of a deployable truss structure had been selected as a placeholder to be replaced by a more mature design during this second study. This has successfully been accomplished in close cooperation with the structure domain. The new concept is based on a fix rectangular truss with a lightweight framework structure covered with MLI (c.f. 3.2.5). This simple solution has the advantage that no deployment mechanism is needed and simplifies the load transfer through the vehicle during launch. Therefore, the transition from the rectangular truss to the cylindrical PL had to be designed using a conical adapter and additional support struts.

The difficulties concerning the equipment of such a platform from the PL via an airlock has been solved by placing the airlock inside the structure and using an opening within the truss to reach the payload with the robot arm. Further, the robot has to be movable around the platform to reach every position of it. This has been accomplished by including a rail-system around the conical connection adapter in the front section of the module to transport the robot arm.

Another advantage of the rectangular truss is the fact, that all required payload interfaces (e.g. electrical, fluids) can easily be connected and wired from the inside of the structure, as there is sufficient available volume for harness and pipes. The mechanical interfaces can be placed on the flat sides with only little support structure needed. The shown mechanical interfaces in Figure 3-9 are placeholders which where adapted from the KIBO JEM EF (c.f. [RD 5]). The arrangement of the payload interfaces on each of the sides provides dedicated viewing directions with 8 positions for zenith and nadir looking experiments, respectively. Additionally, there are another 8 regular and 6 small size payload interfaces on the side looking panels. Depending on the experiments needs, the viewing direction of each payload of course can be adapted within the experiment container design at any time, if e.g. additional payloads with the important zenith and nadir viewing directions are desired on the side panels.

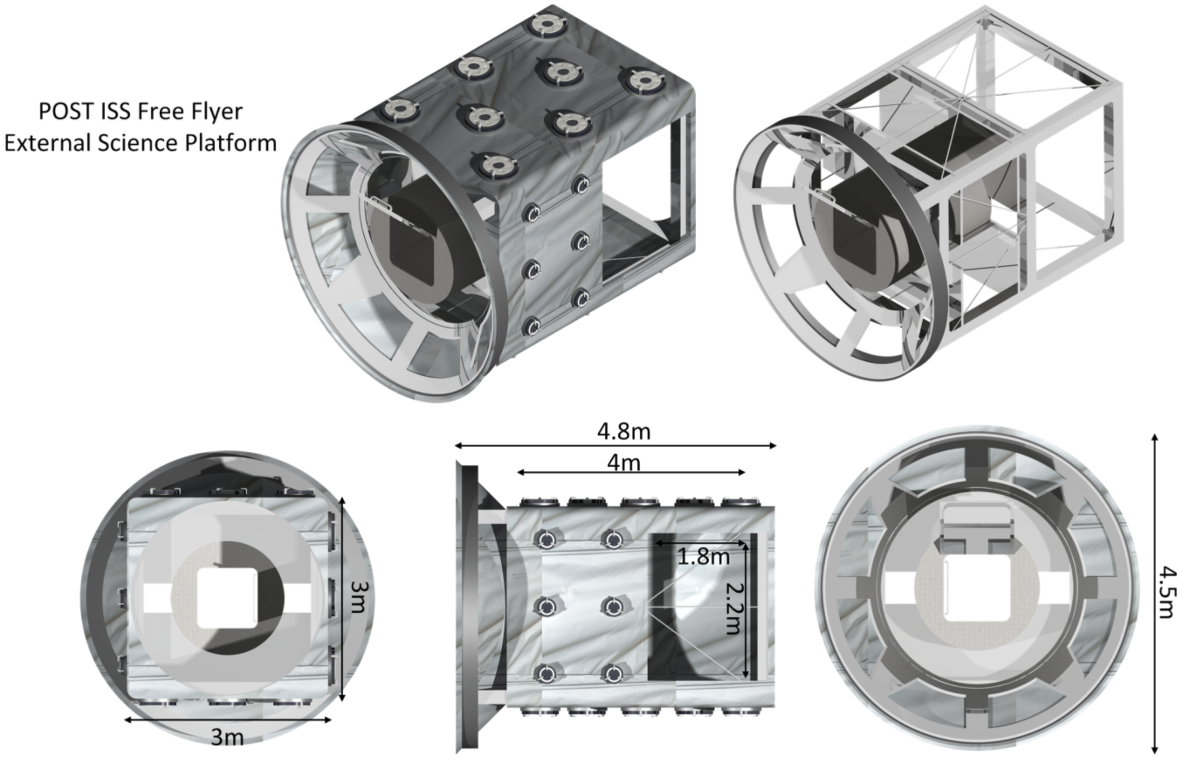


Figure 3-9: Free Flyer ESP (including Airlock) views and outer dimensions

An equipped version of the ESP is shown in Figure 3-10. As most of the envisaged experiments are too big to be mounted on the platform during launch, they have to be placed in orbit using the manipulator.

There is a discrepancy between selected payloads and their associated dimensions and the required IBDM adapter (with 0.8 m passage diameter) of the PL and the derived airlock diameter. To go around that constraint, bigger payloads are planned to be transported directly from a visiting vehicle to the ESP via external manipulators not going through the interior of the station (e.g. HTV and Dragon include unpressurized cargo sections). Another solution for the early phase of the FF might be to store payloads inside the available volume of the External Science Structure during launch in a sophisticated, not yet defined way and extract them from there one after another by the manipulator once in orbit. A more radical alternative to also ensure the capability of subsequent changes of bigger payloads is to change the hatch requirement and replace the IBDM adapter with a yet to be developed adapter with a more wide passage way on the FF and the visiting vehicle.

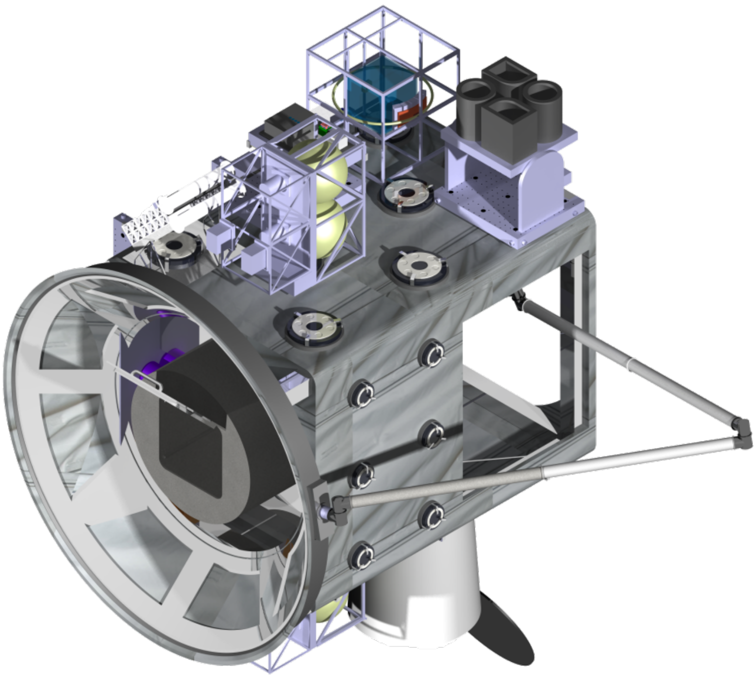


Figure 3-10: Free Flyer ESP equipped with strawman payloads and robot arm

3.2.5. Service Module

The initial design assumed a pressurized SE with a typical barrel shape. During the study, it has been decided that there is no need for the SE to be maintained or accessed and therefore, the pressurized barrel has been replaced by a rectangular lightweight truss structure covered with MLI. By keeping cross section constant between ESP and SE, the two parts can easily be mechanically connected and the loads can be transported through the structure. As for the transition from ESP to PL, the SE requires a conical adapter to connect the rectangular truss with the cylindrical end, which is necessary for the photovoltaic / radiator gimbal ring and the integration of the launch adapter.

One major design driver is the demanded area for photovoltaics and radiators. The first challenge is to accommodate these wings during launch within the available volume. A folding deployment mechanism has been selected and the single solar panels sized to be put close to the rectangular structure. Thereby, the limited width of the single panels leads to a huge length of one solar wing of approx. 28 m each (c.f. Figure 3-12). A more sophisticated solution for the folding mechanism as elaborated in Figure 3-15 may be investigated to reduce the span width if necessary.

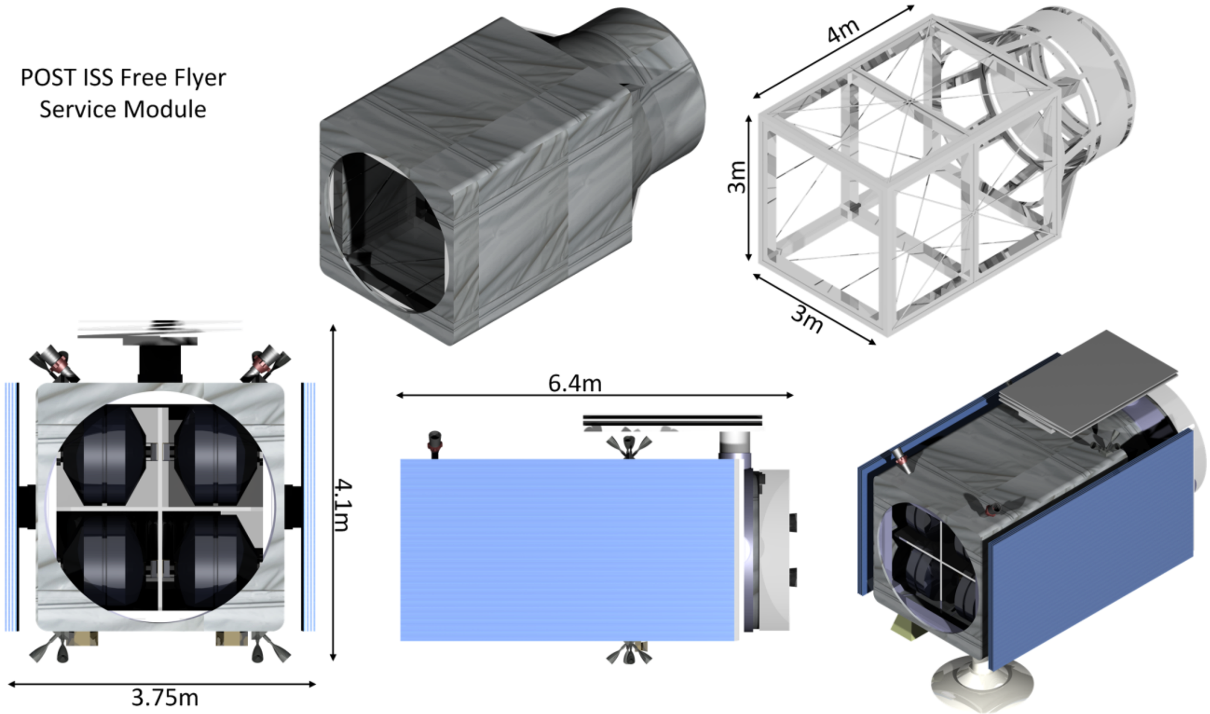


Figure 3-11: Free Flyer Service Module views and outer dimensions in launch configuration

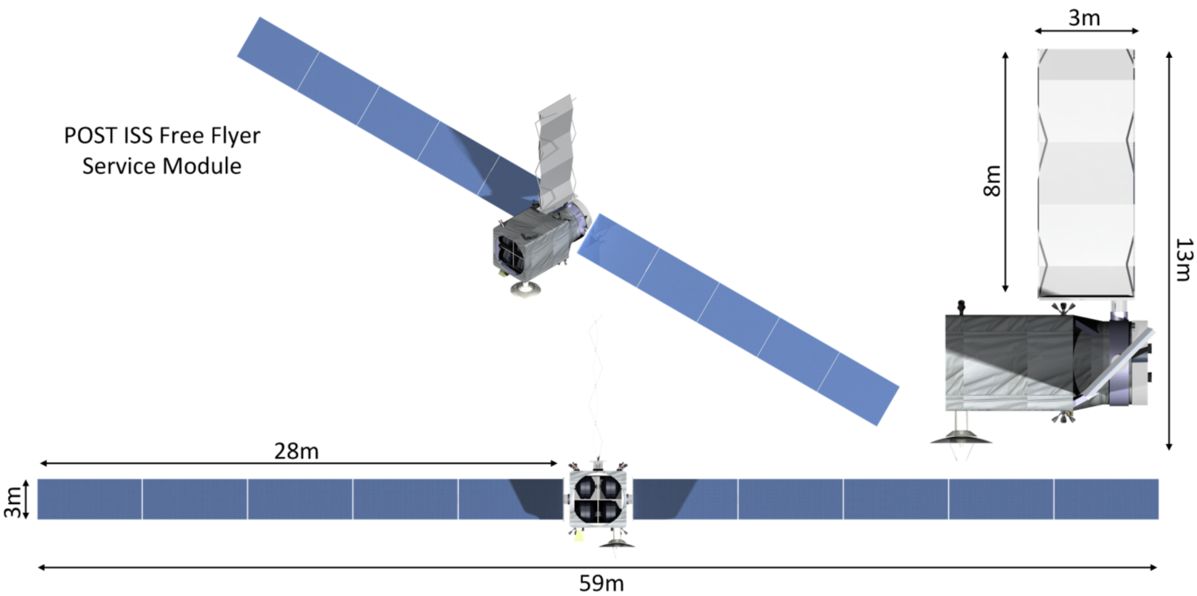


Figure 3-12: Free Flyer Service Module views and outer dimensions in orbit configuration

Additionally, the solar wings have to be controlled in two axes to track the sun in the alpha and beta angle. With respect to the flight direction, the solar angle joints are defined as pictured in Figure 3-13. Following this, the solar wings have to be rotated around the FF's transversal axis to track the sun in its alpha angle (continuous movement with 360°) and around the longitudinal axis for the beta angle (slow oscillation in the range of ± 70°, c.f. Figure 8-5). The solar wings as well as the perpendicular mounted radiator have to rotate around the FF and therefore are integrated on a common bearing ring in the rear of the SE. Therefore, the wings have to have a sufficient clearance to the rectangular structure to avoid interference. A trade had to be done between this clearance and the available volume in stowed configuration during launch. Another open topic is the technology to transfer the generated power and the radiator fluids from the wings into the FF via this bearing ring. Flexible wires / pipes are assumed to be suitable, as the movement around the beta gimbal is slow and limited to a relative small angle range.

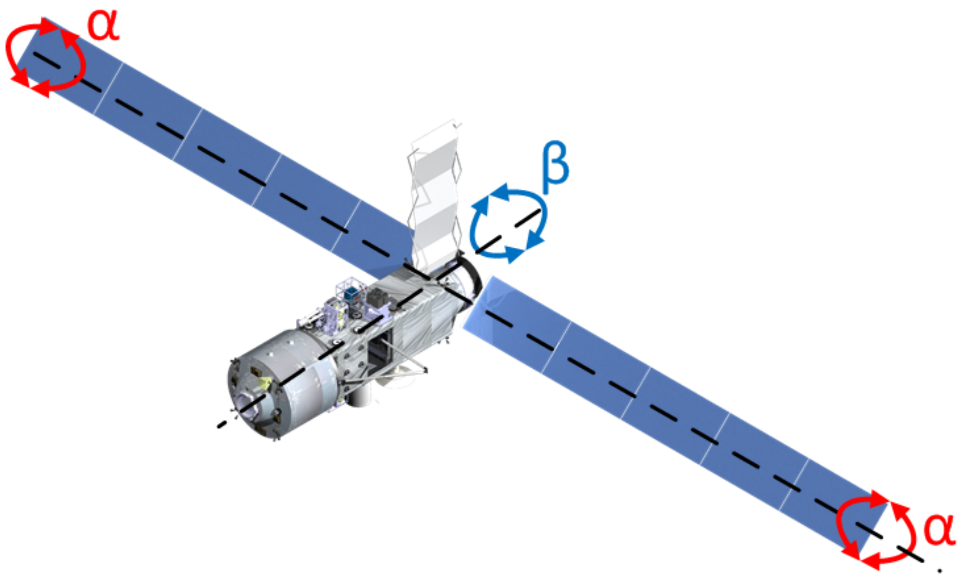


Figure 3-13: Free Flyer solar angle joints

The main components of the subsystems Propulsion, AOCS, Power, DHS and Communication take most of the volume inside the SE, as shown in Figure 3-14. Even though, there are some big components as the batteries, electronic box, CMG assembly and the various propulsion tanks, the SE is able to accommodate all required components thanks to the rectangular structure. One issue which has to be investigated is the dish-type Ku-Band antenna, which would currently stick out the allowed fairing envelope. As there are existing foldable antenna designs [RD 4], the antenna could be stowed during launch and deployed during LEOP, so that this fact has been assumed as minor problem.

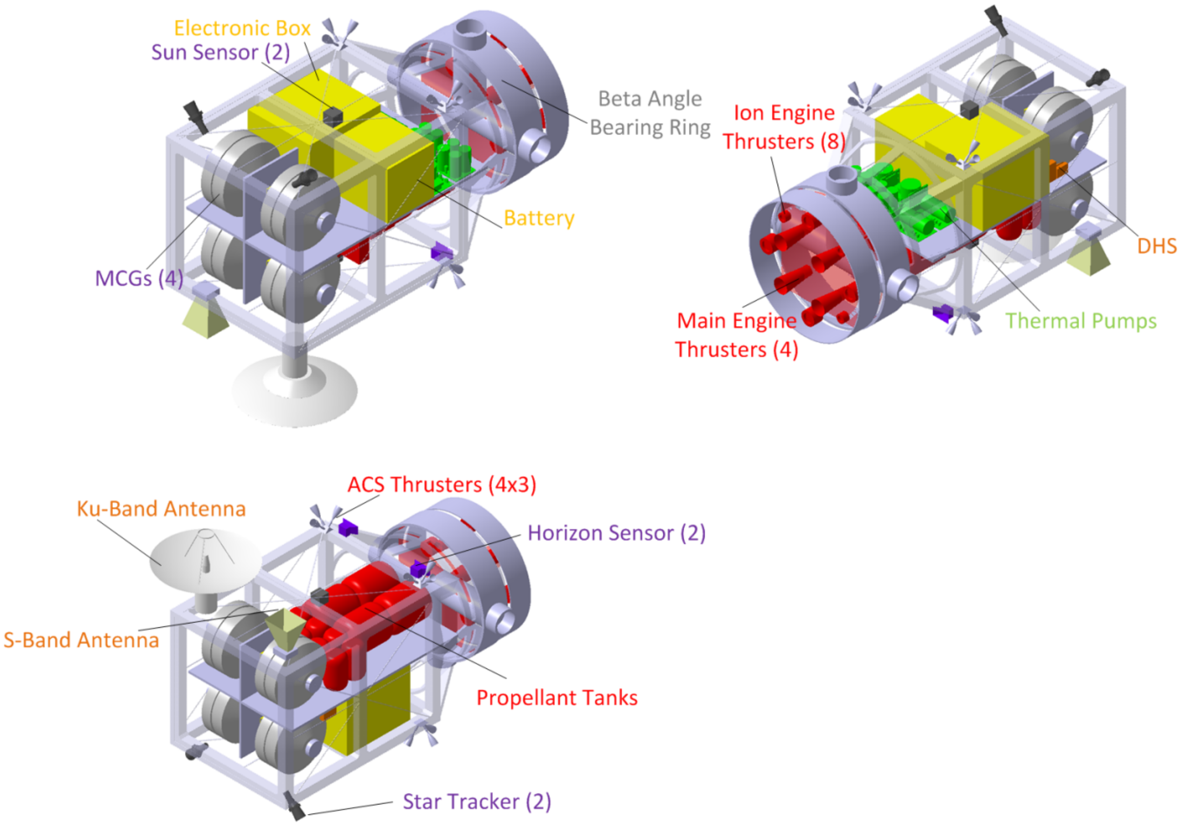


Figure 3-14: Free Flyer Service Module interior view including main components

3.3. To Be Further Studied / Additional Considerations

One of the main points of discussion during and after the study was the design of the FF's solar wings. The striking large span width of the FF using a simple folding mechanism of rectangular solar panels is likely to lead to disturbances and vibrations during operation, which would influence the experiments relying on microgravity environment. Therefore, in the post-processing of the study, several alternative configurations were discussed together with AIRBUS and the options shown in Figure 3-15 were derived. Option 1 shows the concept as included during the study. Option 2 is modelled after existing satellites (mainly used by GEO communication satellites as Astra or Eutelsat), with the advantage of reduced span width [RD 7]. Option 3 is a derivation of that concept. Option 4 assumes two big rectangular solar wings, which would need a very sophisticated folding mechanism to be stowed during launch. Option 5 is a very promising design modelled after the Orbital UltraFlex or MegaFlex solar wings which are flight proven by e.g. the Cygnus cargo vehicle or the Phoenix mars lander. [RD 6]

The circular solar wings shown in Option 5 have been selected as future baseline design in the studies post processing as it has several advantages like a minimal volume in stowed configuration, a minimal span width in deployed configuration as well as a high TRL.

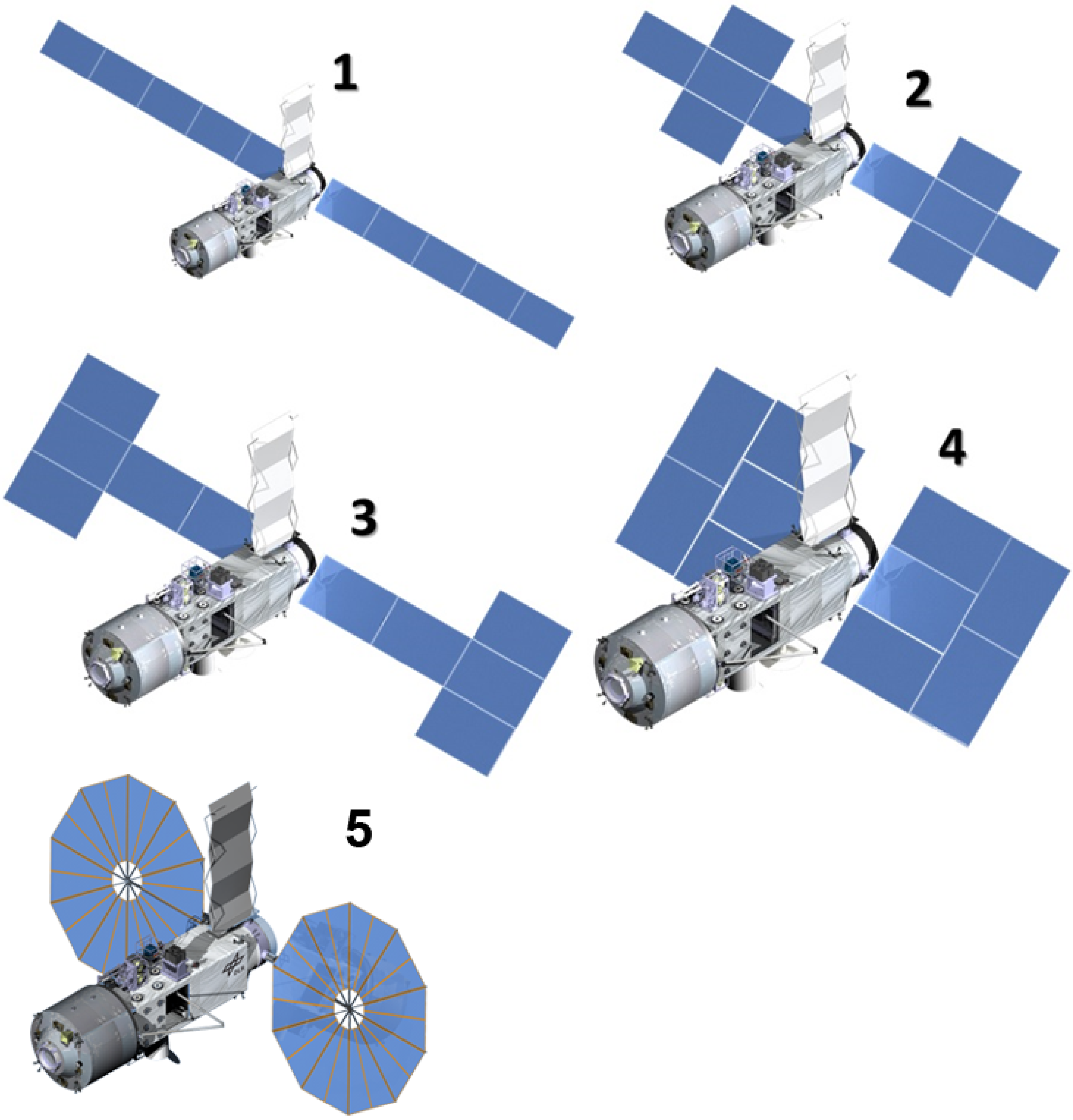


Figure 3-15: Free Flyer solar array design options

Next to this major design change, two minor open points have been described in the chapters above, namely:

- Subsequent equipping of ESP with big payload, does the IBDM has to be replaced by hatch with bigger diameter?
- Prove of beta angle bearing ring feasibility

3.4. Summary

The baseline design of the FF is simplified as much as possible and mainly relying on existing technologies. The main advantage of the redesigned structural concept of the unpressurized parts is the fact that no deployable structure is needed to provide sufficient mounting area for external payloads. The positioning of the airlock inside the truss structure minimizes the reserved volume inside the pressurized part and thereby optimizes the available space for additional experiments. This reduced complexity will likely lead to a reduced development time and to a weight optimization. The place-holder structure introduced in the previous study has successfully been replaced by a consistent, well-thought-out design which fulfils all requirements and design constraints and supports the objective of this experiment platform.

4. Payloads / Science

4.1. Requirements and Design Drivers

- Consider proposals from science community (DLR-workshop May 2014 and CE-study Dec 2014)
- Make reasonable assumptions for mass/power/etc.-requirements where not specified yet by discipline-experts (→ to be confirmed and updated later)
- Include system component "ESP_EOB_MUSES" for Earth Observation instruments (2 large + 2 small) with pointing platform
- Include astronomy place-holders ("ESP_Sol_Observatory" and "ESP_SPOrt") for monitoring the sun and sky survey of microwave background polarisation.

4.2. Baseline Design

4.2.1. Material Science

- PL_MUMS (~ 5 ISPRs):
  - Material design from melt, physics of soft matter
  - Observe processes in real time + RT-Command
  - X-ray radiography and tomography
  - 6 large instruments, 6 smaller self-standing inserts
  - Vol. ~ 8 m³ → ~ 5 ISPRs
  - Total ~ 2000 kg, 3 kW average, 6.5 kW peak, 50% duty cycle, downlink > 100 GByte/day

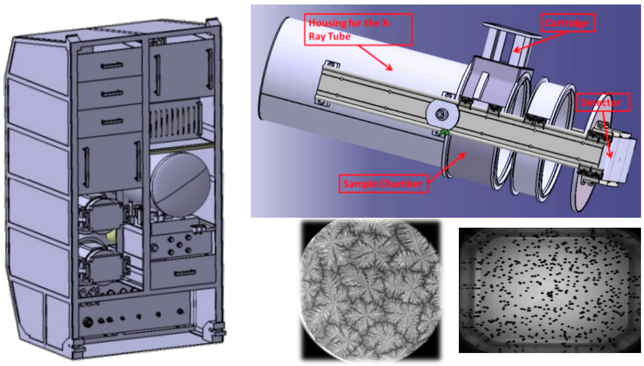


Figure 4-1: Example Equipment for Material Science in the Pressurized Laboratory

4.2.2. Atmospheric Physics (passive)

- ESP_Atmo_passive_Spektrometer
  - UV-VIS-NIR-SWIR spectrometer: up to 5 different modules (50x30x70 cm³)
  - Total 250 kg, 0.5 kW average, 40% duty cycle, downlink 20-200 Mbits/s/module

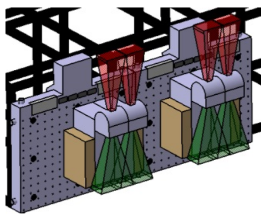


Figure 4-2: UV-VIS-NIR-SWIR Spectrometer on ESP

4.2.3. Atmospheric Physics (active)

- ESP_Lidar_Observatorium
  - Test of new Lidar-techniques and instruments
  - Active measurements of atmospheric trace elements
  - Nadir telescope ≥ 1 m aperture receiving back-scattered laser-light
  - Exchangeable transmitter + receiver module (~2 years lifetime)
  - Total: ~ 400 kg, 3.5 kW + 3.5 kW thermal, 50% dutycycle, downlink 4 Mbit/s

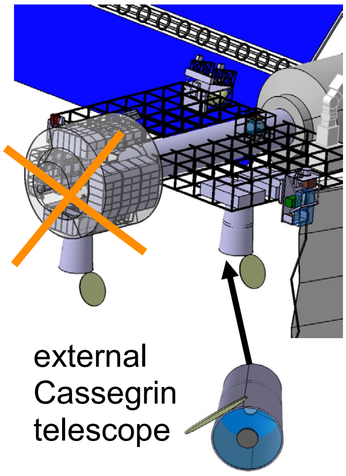


Figure 4-3: Atmospheric Physics Lidar on ESP

4.2.4. Earth Observation

- ESP_GPoptEO
  - 2 telescopes UV-VIS-NIR-SWIR-MWIR-LWIR (2°+ 20° FOV) each: 2x1.5x1m³, 250 kg
  - 3 standard chambers (temp. controlled) for 6 instruments each: 1.4x0.7x0.5 m³, 100 kg
  - Total mass 1500kg, 1kW, 2kW peak, 40% duty cycle,
    downlink > 3.3 TByte/day

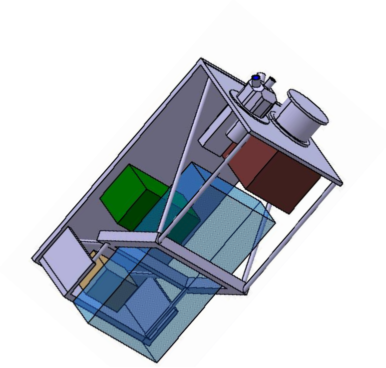


Figure 4-4: Earth observation telescope on ESP

- ESP_EOB_MUSES
  - Pointing platform MUSES: 120 cm x 120 cm, 300 kg, 0.5 kW
  - 2 large instr., each < 100 kg, 55x55x90 cm³, 0.3 kW
  - 2 small instr., each < 50kg, 35x35x90 cm³, 0.2 kW
  - Total: 600 kg, 1 kW, 40% duty cycle, downlink ~200 GByte/day

Figure 4-5: Pointing platform MUSES on ESP

4.2.5. Astrobiology

- ESP_PlumeSIM
    - Instrument test and sample analysis (biosignatures) under simulated icy-moon conditions (Plume simulator: H_2O +… injector etc.)
    - Spectrometer (Raman)
    - Total: ~150 kg, 0.3 kW, 5% duty cycle, downlink rate small

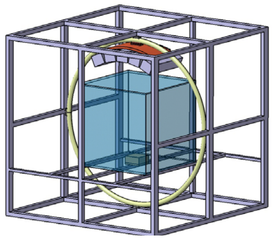


Figure 4-6: Plume simulator on ESP

- ESP_EXPOSE_platform
    - Micro-ecosystems under space conditions using sunlight, different spectrometers
    - Total: ~150 kg, 0.3 kW, 50% duty cycle, downlink rate small

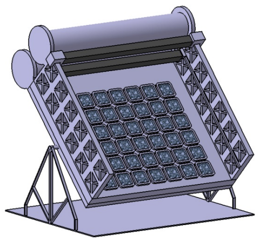


Figure 4-7: EXPOSE platform on ESP

4.2.6. Technology Demo

- ESP_Elektrisches_TW
    - Electric propulsion testplatform
    - Eventually divided in 2 parts for thrust compensation
    - Total ~ 600 kg, 6 kW, duty cycle 50%, downlink rate small

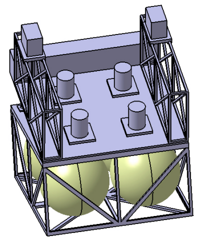


Figure 4-8: Electrical propulsion as technology demonstrator on ESP

4.2.7. Astronomy

- ESP_SPOrt
    - Sky Polarization Observatory → sky survey of microwave background
    - Origin: proposal as Columbus external P/L,
    - Total: ~ 500kg, 0.8 kW (closed loop cryocooler), 100% duty cycle, downlink ~200 GByte/day
- ESP_Sol_Observatory
    - Solar Observatory → pointing platform required (300kg)
    - Total ~ 800kg, 0. 8kW, 20% duty cycle, ~200 GByte/day

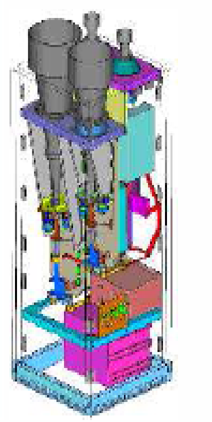


Figure 4-9: Astronomy payload on ESP

4.3. Payload Budgets

4.3.1. List of Equipment

Table 4-1: Mass budget of the payloads.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
ESP__EXPOSE_platform	150.00	20.00	30.00	180.00	2.16
ESP_Atmo_passiv_Spektrometer	250.00	20.00	50.00	300.00	3.60
ESP_Elektrisches_TW	600.00	20.00	120.00	720.00	8.63
ESP_EOB_MUSES	600.00	20.00	120.00	720.00	8.63
ESP_GPoptEO	1500.00	20.00	300.00	1800.00	21.58
ESP_Lidar_Observatorium	400.00	20.00	80.00	480.00	5.76
ESP_PlumeSIM	150.00	20.00	30.00	180.00	2.16
ESP_Sol_Observatory	800.00	20.00	160.00	960.00	11.51
ESP_SPOrt	500.00	20.00	100.00	600.00	7.19
PL_MUMS	2000.00	20.00	400.00	2400.00	28.78

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	6950.00			8340.00	

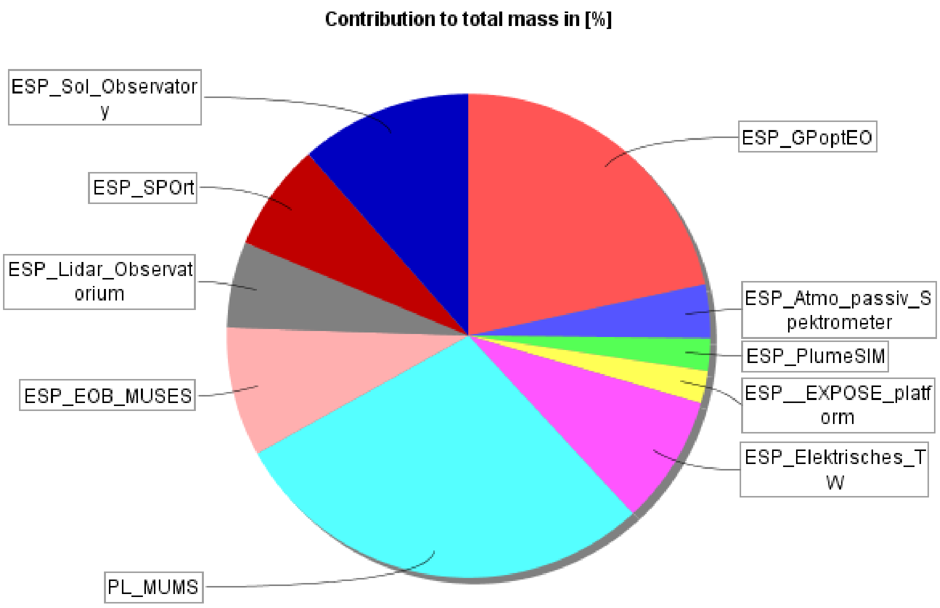


Figure 4-10: Pie chart of mass distribution of all payloads

4.3.2. Power Budget

Table 4-2: Power budget of the payloads.

System Component	Parameter	Unit	Standard Mode	Survival Mode
Payload_Science	power_avg_wMargin	Watt	10069.200	0.000
ESP_GPoptEO	power_avg_wMargin	Watt	487.200	0.000
ESP_Atmo_passiv_Spektromet	power_avg_wMargin	Watt	240.000	0.000
ESP_Elektrisches_TW	power_avg_wMargin	Watt	3600.000	0.000
ESP_PlumeSIM	power_avg_wMargin	Watt	18.000	0.000
ESP__EXPOSE_platform	power_avg_wMargin	Watt	180.000	0.000
ESP_EOB_MUSES	power_avg_wMargin	Watt	480.000	0.000
ESP_Lidar_Observatorium	power_avg_wMargin	Watt	2100.000	0.000
PL_MUMS	power_avg_wMargin	Watt	1812.000	0.000
ESP_SPOrt	power_avg_wMargin	Watt	960.000	0.000
ESP_Sol_Observatory	power_avg_wMargin	Watt	192.000	0.000
Payload_Science	power_energy_wMargin	Joule	73078225920...	0.000

4.3.3. Mode Dependencies

All payloads can be deactivated during survival mode; there only very small power may be required for temperature control and cooling of samples.

4.3.4. Data Rate & Volume

Most required data rates & volumes are standard and comparable with ISS capabilities; some demanding requirements may derive from future Earth observation payloads (e.g. more than a few TByte/day downlink due to very high resolution observations)

4.4. Re-supply Items / Return Capability

- **PL_MUMS**: regular sample/insert exchange
  → upload/download ~ 30 kg per 3 months
- **ESP_Atmo_passiv_Spektrometer**: exchange of contaminated modules
  → upload/download ~ 50 kg per year (esp. UV-modules, download for inspection and on-ground calibration of degradation)
- **ESP_Lidar_Observatorium**: exchange of transmitter/receiver modules
  → upload ~ 100 kg per 2 years (download desirable but not necessary)
- **ESP_GPoptEO**: exchange of instruments (lifetimes 0.5 to 3 years)
  → upload/download ~ 200 kg per year
- **ESP_EOB_MUSES**: exchange of 2 large and 2 small instruments on pointing platform
  → upload/download ~ 300 kg after 3-4 years
- **ESP_PlumeSIM**: exchange of glass container
  → upload/download ~ 20 kg per 3 month

- **ESP_EXPOSE_platform**: exchange of container trays
  → upload/download ~ 20 kg per year
- **ESP_Elektrisches_TW**: exchange of complete propulsion module
  → upload/download ~ 600 kg after 3-4 years

4.5. To Be Further Studied / Additional Considerations

- Check and confirm or modify numbers with experts
- Fill placeholders with reasonable ideas and numbers
- Select (straw man) payloads which could be (must be?) launched initially with the Free Flyer (→ to stay within required total launch mass limit)
- Define procedure for subsequent transportation and placement of potential payloads on platform after Free Flyer launch (e.g. with crew support via pressurized volume and airlock plus robotic arm or using unpressurized cargo ship and robotic transfer via 'free space' for larger payloads - requiring an extra/different robotic arm)
- Investigate restrictions on potential payloads from airlock dimensions
- Consider further astronomy straw man payloads, e.g.:
  - **SVOM-Like** (Space Variable Objects Monitor – Swift successor) X-ray and GAMMA-ray bursts detection → pointing platform required
  - **Exoplanet Search** (~1m telescope) → pointing platform required

4.6. Summary / Comparison to ISS

- Some of the Earth observation and astronomy payloads require dedicated pointing platforms (if the mentioned MUSES platform - under construction at Teledyne Brown Engineering, Inc. TBE (USA) for ISS applications - is sufficient or not is still unclear)
- Most experiments have finite operation-time of about one to several years; thereafter they have to be removed/exchanged
- A lot of resupply items are required for some of the experiments during operation
- The total mass of all proposed straw man payloads for the Free Flyer is more than 8000 kg. Probably they cannot all together be accommodated on the initial launch configuration of the Free Flyer and must be transported with later resupply flights/launches. This may cause some problems, if the dimensions of the hardware are large (on the order of about 1 meter or more), because the standard dimensions of the airlock for pressurized payload transfer to the Free Flyer platform are Ø= 80 cm.

- Power requirements can be managed via time sharing because most payloads have only duty cycles of typically 50 % or less
- Contrary to the situation on ISS, future material science experiments will be fully automatic and/or monitored on ground in real time with transmission of in-situ diagnostics (video surveillance and tele-science); crew activities may only be required from time to time (typically every 3-6 month) for exchange of experiments and samples/inserts. The same holds true for earth observation payloads, which may be relatively short lived or could be typically of experimental type (contrary to operational), where the crew is needed for exchange of damaged or contaminated modules/instruments (typically every few years).

5. Crew Workstation

5.1. Requirements and Design Drivers

- Facilitate a comfortable and effective working position for the crew to work on payloads for sample exchange and/or reconfiguration tasks
- Position and hold payload or workpiece in place
- Provide close by a subset of standard tools for ease and speed of operations
- Provide room and fixation capability for samples, spare parts, tools and procedures

5.2. Baseline Design

- A lightweight plate which can be attached to the front of an ISPR onto the handrails and takes up about a fourth of the ISPR front
- Provides a set of attachment provisions for bigger payloads and for several smaller parts on the circumference realized with Velcro, Velcro straps of various length and bungees being interchangeable
- Includes a tool-shed for a subset of commonly used tools
- Includes a vice with a wide span for exact fixation of parts of various sizes
- Includes a battery operated movable LED light on a flexible arm
- The weight is about 15 kg
- No power consumption

5.3. Re-Supply Items

- Bungees and Velcro straps of various length
- Batteries for the LED lamp

5.4. To Be Further Studied / Additional Considerations

- The specific and detailed design of this Crew Workstation

5.5. Summary / Comparison to ISS

- Debriefs of ISS Expedition Crews always report about the huge overhead for setting up the worksite and gathering the tools and spare parts.
- This Crew Workstation combines a basic set of utilities for ease and speed of use.

6. ECLSS

6.1. Requirements and Design Drivers

- Support the safe and continuous work of two crew members inside the PL of the Free Flyer
- The air composition and humidity is controlled via the Base Station or any other visiting vehicle when the FF is crewed
- The PL should provide adequate lighting, ventilation and air temperature control in its common air volume
- The PL should be protected against over pressurisation
- The PL should have a ‚Fire Detection and Suppression System' in the common air volume
- The PL should provide adequate ‚Emergency Equipment' for two crew members

6.2. Baseline Design

- The Emergency Equipment consists of: PBA (mask) with Oxygen bottle, Fire extinguisher, goggles and a CSA-CP (Compound Specific Analyser for Combustion Products) for each crew
- The PPRA (Positive Pressure Relieve Assembly) protects against over pressurization
- The Ventilation provides guided airflow in the PL common air volume. Two fans are located in two parallel ducts which can be closed individually to create functional redundancy and additional variability of airflow
- Temperature control is provided by a heat exchanger on a cold plate in the common air duct and two heaters with associated thermostat electronics in the two separated fan air ducts
- Lighting is provided by six LED assemblies. Each one is dimmable and can be switched on independently. Three are mounted on top of three neighbouring racks and also three on the opposite side of these racks.
- The Fire Detection and Suppression System consists of two redundant Fire and Smoke Detectors inside the PL common air volume and of two redundant high pressure N2 tanks which are located outside on the hull right beside the external airlock. The actual fire suppression can be achieved during unmanned FF phase by venting atmosphere overboard and resupplying N_2 in a staggered way: the associated change in PL pressure should be limited to -/+ 20% for each individual step.

6.3. Options and Trades

- Design is for two continuously working crew members inside the PL
- Reduce Fire Suppression (N_2 tank) from two to one (only for FF mode)
- No Emergency Equipment - take Emergency Equipment (one or both) from Base Station to the PL during crewed phases
- Reduce continuous presence during crewed phases to one crew member

6.4. Mass and Power Budget

- Rough estimates

6.4.1. List of Equipment

Table 6-1: Mass budget of the ECLSS.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Emergency_Equipment	28.00	5.00	1.40	29.40	17.17
Fire_Detection_Suppression	76.00	10.00	7.60	83.60	48.82
Lighting	20.40	10.00	2.04	22.44	13.10
Positive_Pressure_Relieve_Valve	4.00	5.00	0.20	4.20	2.45
Temperature_Control	8.00	20.00	1.60	9.60	5.61
Ventilation	20.00	10.00	2.00	22.00	12.85

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	156.40			171.24	

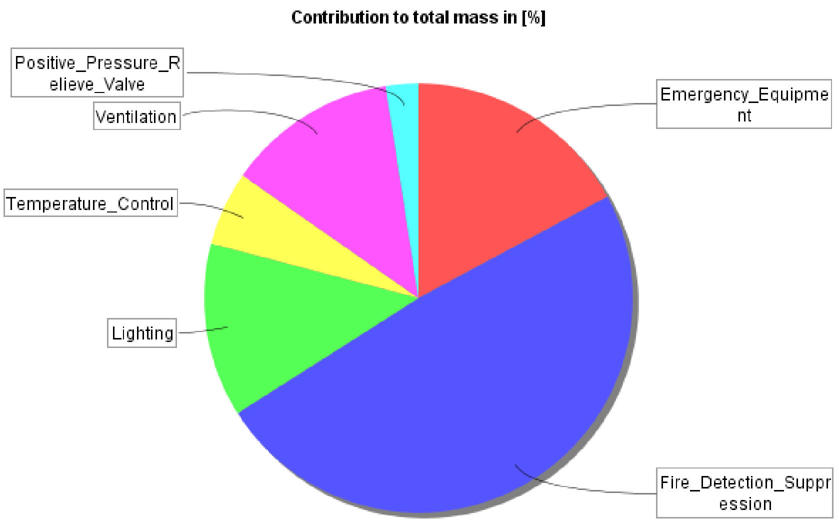


Figure 6-1: Pie chart of mass distribution of the ECLSS

6.4.2. Power Budget

Table 6-2: Power budget of the ECLSS.

System Component	Parameter	Unit	Standard Mode	Survival Mode
ECLSS	power_avg_wMargin	Watt	351.321	0.000
Emergency_Equipment	power_avg_wMargin	Watt	0.000	0.000
Fire_Detection_Suppression	power_avg_wMargin	Watt	16.500	0.000
Lighting	power_avg_wMargin	Watt	19.800	0.000
Positive_Pressure_Relieve_Valv	power_avg_wMargin	Watt	0.021	0.000
Temperature_Control	power_avg_wMargin	Watt	150.000	0.000
Ventilation	power_avg_wMargin	Watt	165.000	0.000
ECLSS	power_energy_wMargin	Joule	25497472896...	0.000

6.4.3. Mode dependencies

During Survival Mode no power generation and only minimized power consumption is assumed for the PL systems. As a consequence also heat sources are minimized. Thus for a period of two days ECLSS can be shut down in Survival Mode.

6.5. Re-Supply Items

- None for nominal operations

6.6. To Be Further Studied / Additional Considerations

- Is the temperature control sufficient for the actual payload situation? (how much dissipated power into the common air volume)
- Is active Fire Suppression during FF phase really needed in the common air volume?

6.7. Summary / Comparison to ISS

- Less capable - but adequate for short periods (two weeks) with the Base Station or another suitable visiting vehicle docked to the PL
- No exact temperature control. Set temperature only approximated within a wider temperature range (10 to 30 Degree Celsius)

7. Venting Systems

7.1. Requirements and Design Drivers

- 15 years operations
- Venting is provided for four internal payload locations
- The function is zero failure tolerant (FT) for function
- The venting is 2 FT for safety (loss of atmosphere)
- The venting system can be used by one payload at the time (to exclude cross contamination)

7.2. Baseline Design

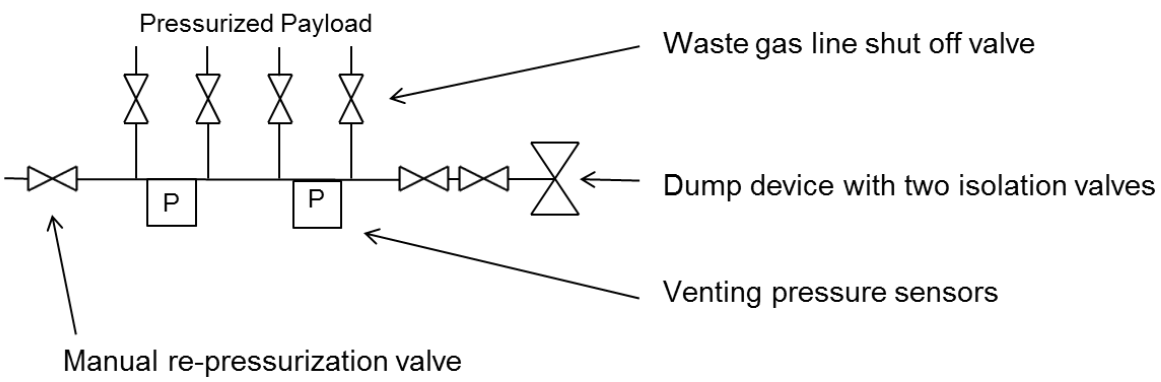


Figure 7-1: Scheme of the Venting Systems baseline design

7.3. Options and Trades

- Delete the venting function as standard service for payload, which could reduce flexibility for material science payload

7.4. Mass and Power Budget

Mass and power data is derived from ISS hardware data.

7.4.1. List of Equipment

- None-thrust dump device with two isolation valves
- Waste gas line shut off valve per payload position to isolate the payload rack from the venting system
- Two venting pressure sensors (for redundancy and consistency) to measure the current internal pressure for the venting system. This information is used to control the waste gas line shut off valve in front of the payload and for the detection of safe condition in case of payload operations
- Vent system re-pressurization valve to provide the ambient pressure inside the vent system for the preparation of a payload exchange and leakage detection

Table 7-1: Mass budget of the Venting Systems.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
PL_Dump_Device	1.90	5.00	0.10	2.00	4.21
PL_Repressurization_Valve	1.00	5.00	0.05	1.05	2.22
PL_Venting_Lines	30.00	20.00	6.00	36.00	76.05
PL_Venting_Pressure_Sensor	1.50	5.00	0.07	1.57	3.33
PL_Waste_Gas_Line_Shut_off_Valve	6.40	5.00	0.32	6.72	14.20

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	40.80			47.34	

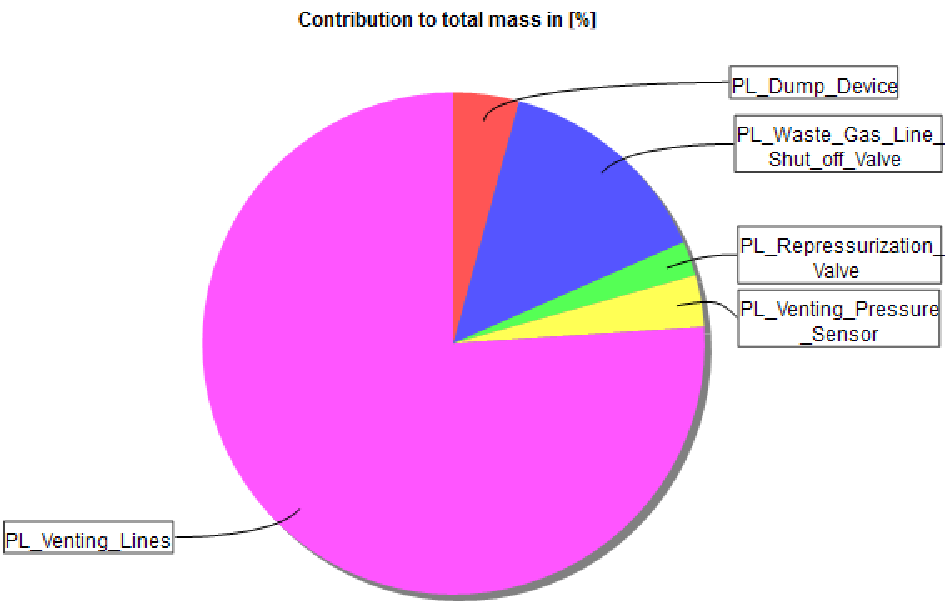


Figure 7-2: Pie chart of mass distribution of the ECLSS

7.4.2. Power Budget

Table 7-2: Power budget of the ECLSS.

System Component	Parameter	Unit	Standard Mode	Survival Mode
Venting_Systems	power_avg_wMargin	Watt	1.578	0.002
PL_Dump_Device	power_avg_wMargin	Watt	0.002	0.000
PL_Repressurization_Valve	power_avg_wMargin	Watt	0.002	0.000
PL_Venting_Lines	power_avg_wMargin	Watt	0.000	0.000
PL_Venting_Pressure_Sensor	power_avg_wMargin	Watt	1.575	0.000
PL_Waste_Gas_Line_Shut_off_	power_avg_wMargin	Watt	0.000	0.002
Venting_Systems	power_energy_wMargin	Joule	114535814.400	272.160

7.4.3. Mode dependencies

The venting system is beside the pressure sensors most of the time passive. Only in case of payload reconfiguration the valves are operated for a short time of several seconds. The venting mode is not needed for the survival mode and can be completely deactivated.

7.5. Re-Supply Items

- None for nominal operations

7.6. To Be Further Studied / Additional Considerations

- Confirmation if payload can operate with a single vacuum/venting system (separation of venting and vacuum application, higher cleanliness level for vacuum)

7.7. Summary / Comparison to ISS

- Simplified system, ISS provides separate venting and vacuum systems

8. Mission Analysis

8.1. Orbit Requirements and Design Drivers

- Target orbit: Inclination 51.6°; Altitude=400 km
- Solar Flux Fp=150…300
- Nadir pointing with velocity constraint
- (Inertial pointing for certain periods)

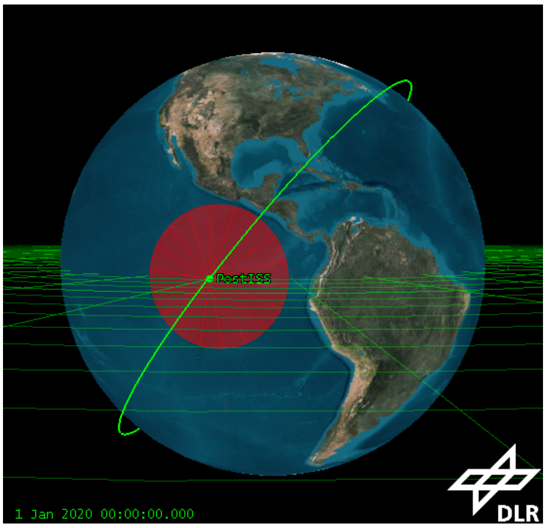


Figure 8-1: Post-ISS orbit visualization

8.2. Baseline Orbit

For the Free-Flyer different attitude profiles are required: Either nadir fixed with velocity constraint or inertial pointing (e.g. towards celestial objects) or inertial fixed in all axes. Therefore beside the main nadir profile in the section 8.3 also other orientations are regarded.

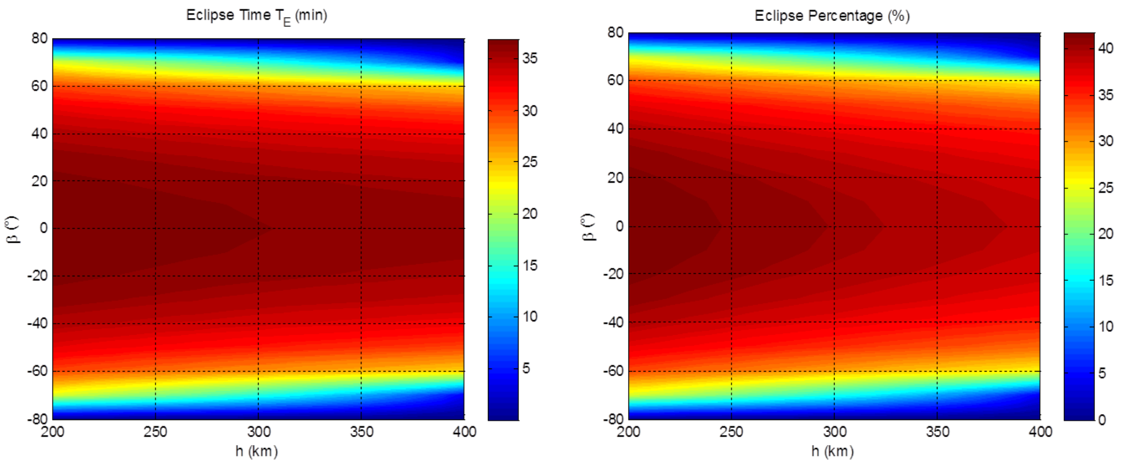


Figure 8-2: Eclipse times depending on beta-angle and altitude (left: in min.; right: in %)

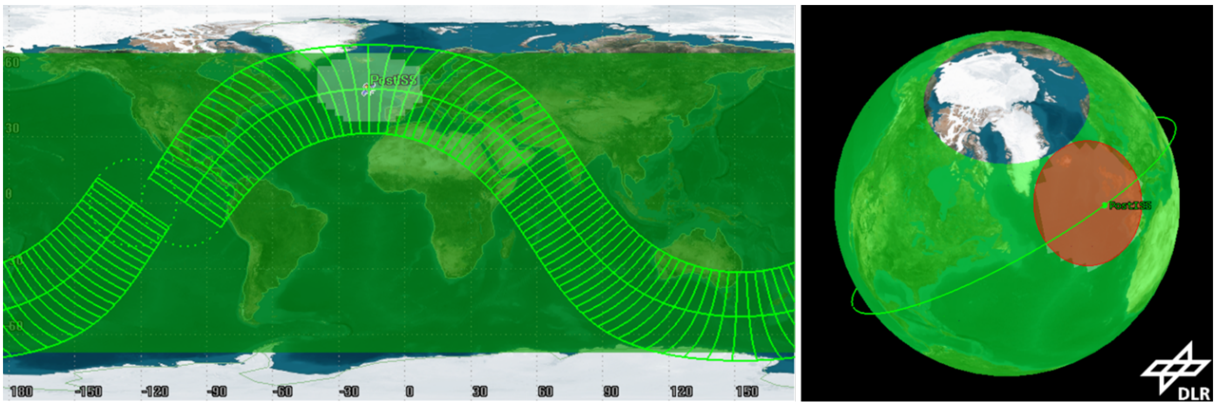


Figure 8-3: Coverage of the baseline orbit (51.6°incl.; 5° elevation)

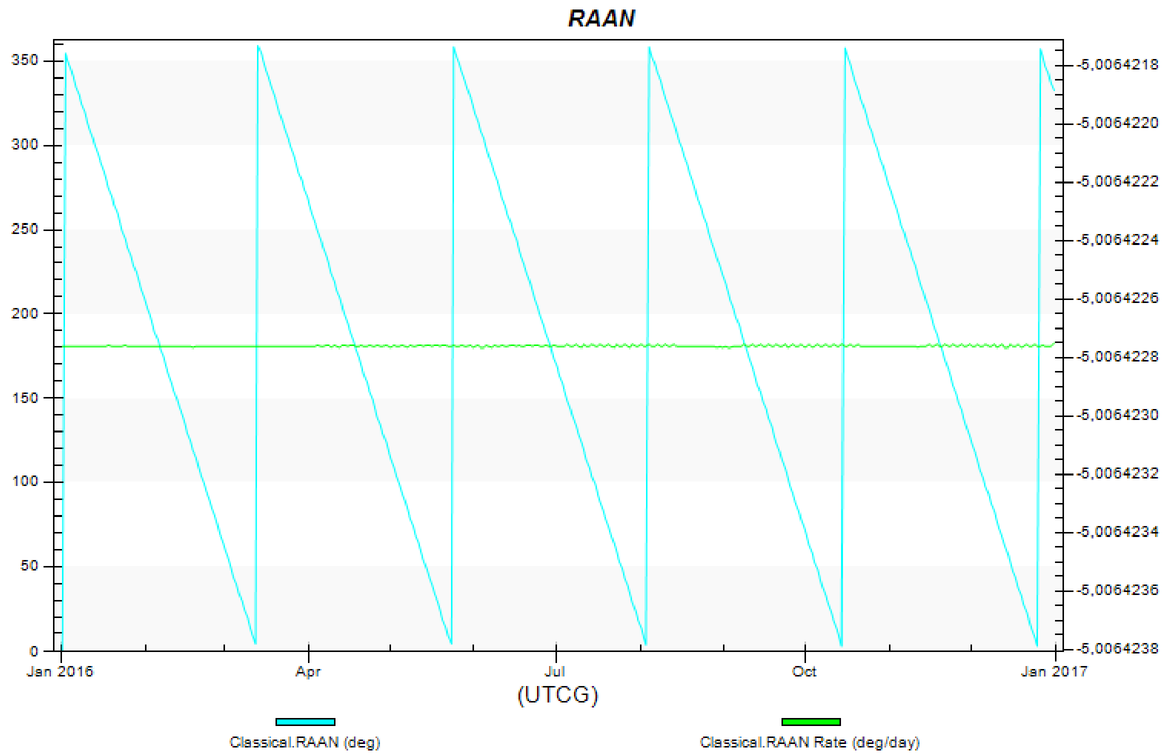


Figure 8-4: RAAN and RAAN drift rate for the baseline orbit.

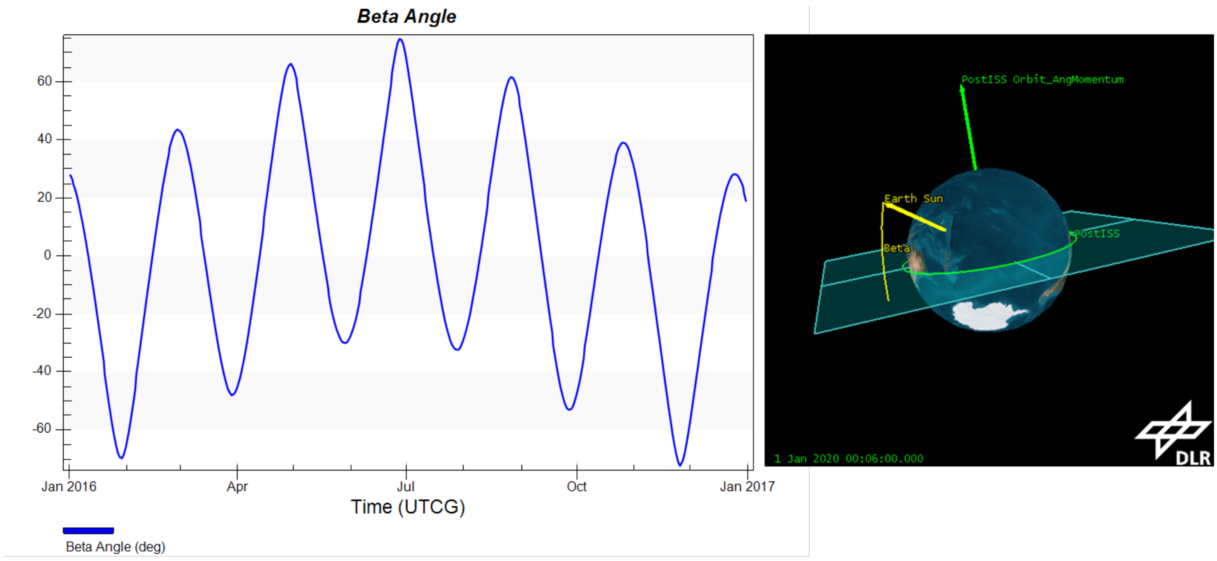


Figure 8-5: Beta-angle of the baseline orbit in deg. over one year.

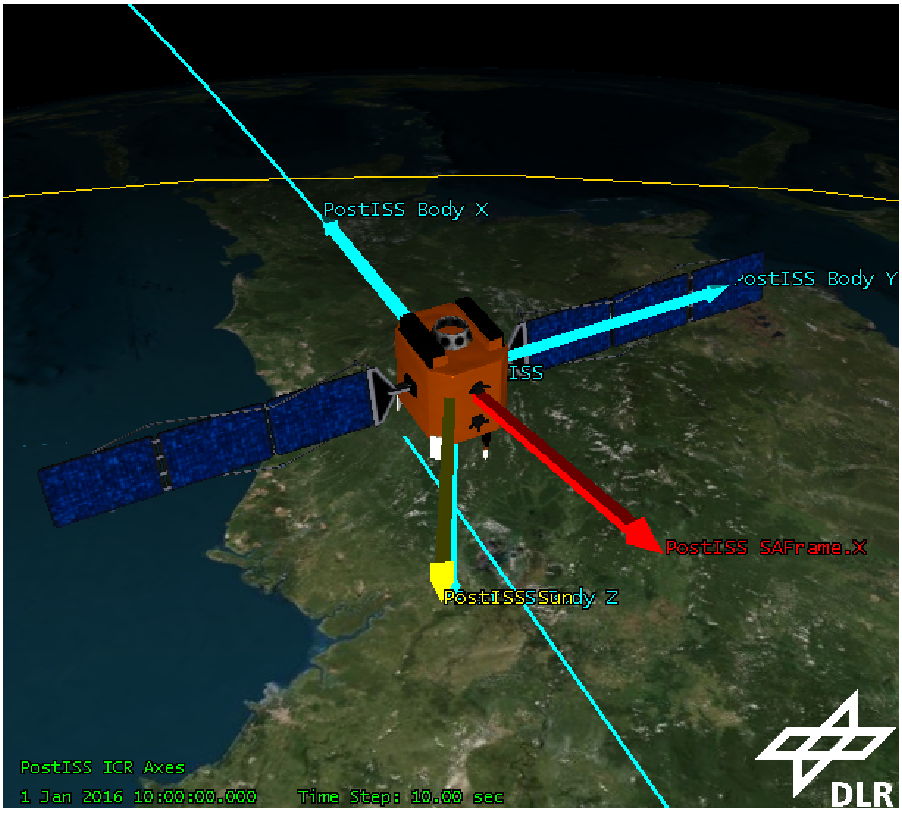


Figure 8-6: Baseline attitude (z=Nadir; x constraint to velocity).

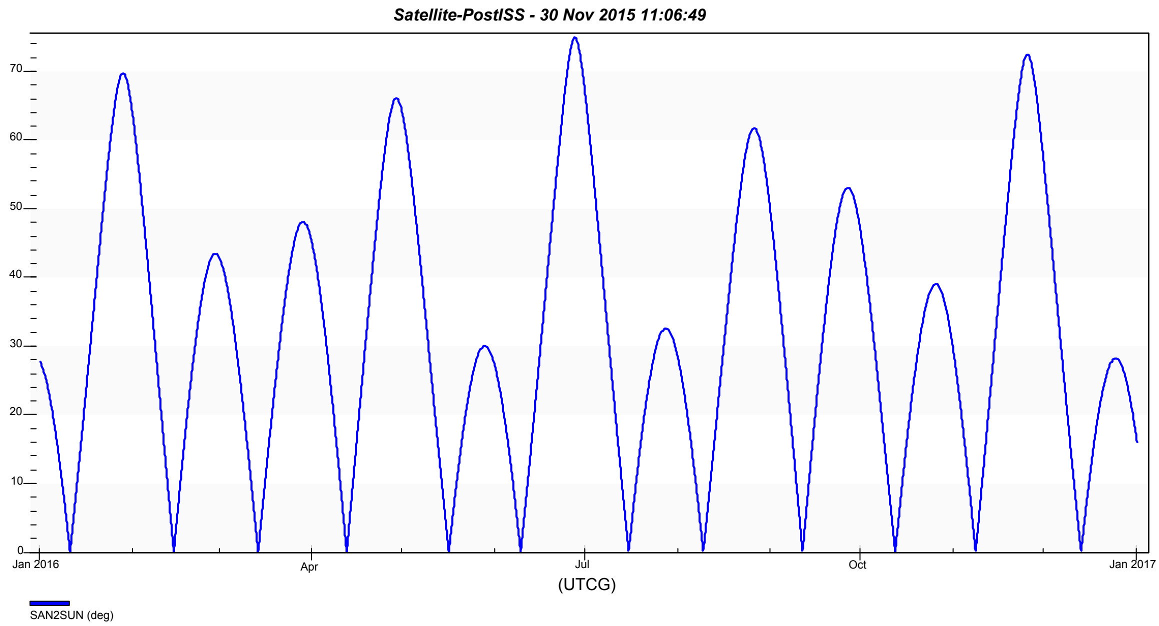


Figure 8-7: Sun angle vs. Free Flyer baseline attitude.

8.3. Orbit Options and Trades

In the following the solar-area-normal to Sun angles and the velocity to the Free Flyer's X-axis angle are given for one year.

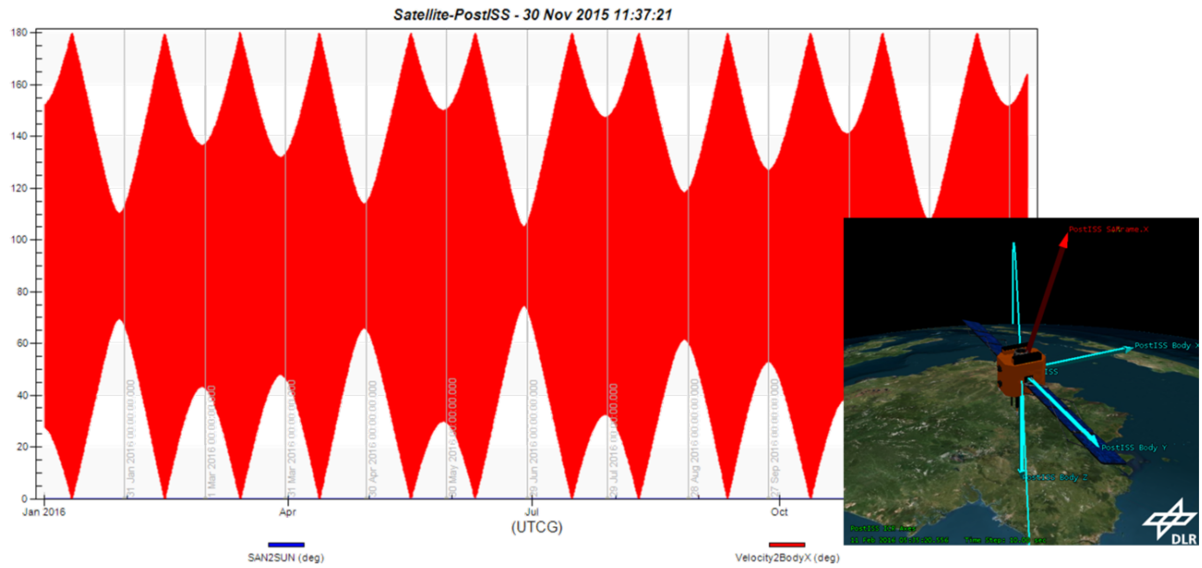


Figure 8-8: z-nadir pointing with x constraint to Sun.

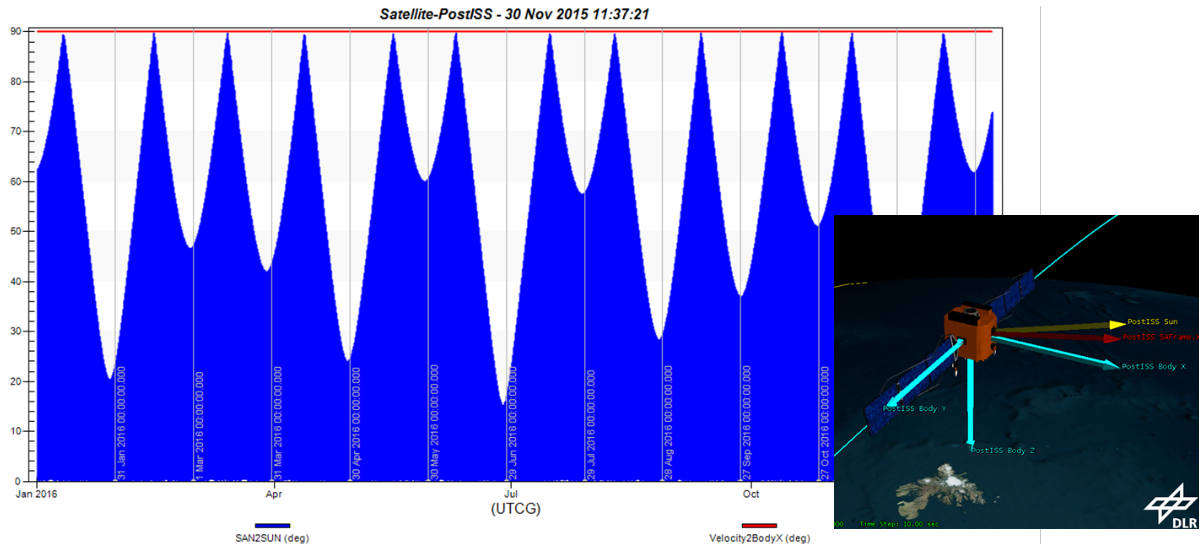


Figure 8-9: z-nadir pointing with y constraint to velocity.

8.4. Manoeuvre Requirements and Design Drivers

- On-orbit mass is 25 t
- Constant orbit (400 km, 51.6 deg)
- Free-Flyer is the leading spacecraft; base station is the trailing spacecraft
- Standard orientation is nadir for P/L → drag torque has to be countered

Rendezvous and Docking:

- Phasing is performed by Base Station in combination with its orbit raising manoeuvres (ORM)
- Docking performed within six hours (one ground control shift)
- Safe distance between spacecraft is 100 km
- Final approach based on ATV
- Rendezvous and docking (RVD) manoeuvres performed using chemical thrusters
- RVD cycle: 12 weeks of free flying plus 2 weeks docked to Base Station

8.5. Manoeuvre Baseline

- Orbit raising manoeuvres:
    - Using electrical engines (RIT 10 EVO and RIT 22)
    - No orbit raising but thrust control (10.1 – 129 mN) to achieve constant orbit height
    - 656 kg Xenon required for 15 yrs (343 kg for smaller solar panels and 10 yrs MLT)

- Debris avoidance manoeuvres (DAM):
    - Executed by chemical propulsion system
    - 2.5 DAMs per year (comparable to ISS) requires about 184 kg propellant total for 15 yrs

- Rendezvous and Docking:
    - Docking performed within six hours (one ground control shift)
    - Starting point 100 km from Base Station
    - Three orbit hoppings to arrive at 3500 m from station
    - ATV approach from then onward
    - Approx. 505 kg of chemical propellant per docking (biggest demand during close proximity operations)
    - 30.13 t propellant over MLT of 15 years

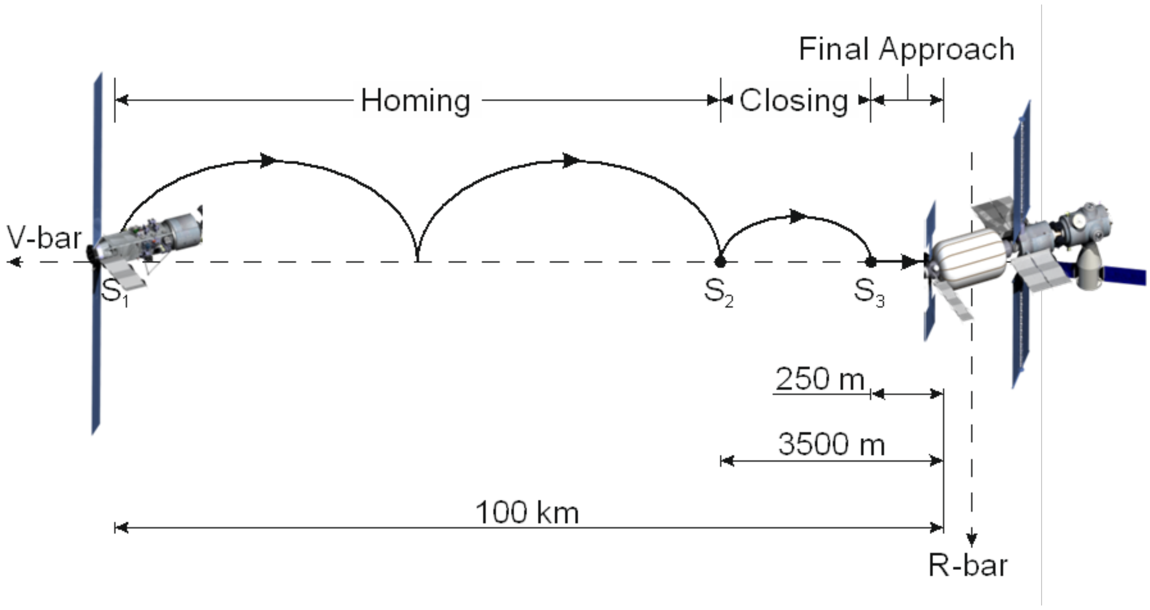


Figure 8-10: Formation Flying and Rendezvous scenario.

8.6. Manoeuvre Options and Trades

- ORM using chemical thrusters:
  - One manoeuvre every 37 to 1615 days
  - Propellant requirement between 20.8 and 59.5 kg per manoeuvre

8.7. To Be Further Studied / Additional Considerations

- Close proximity operations for RVD need to be further studied
- Formation Flying concept, if Base Station uses different/periodical orbit raising strategy

8.8. Summary

The Free-Flyer's orbit is a continuation of the well-known ISS orbit. The selection of a hybrid propulsion system (electrical for drag compensation and chemical for rendezvous) leads to a propellant saving of up to 14% in comparison to a purely chemical approach. Application of continuous thrust for the orbit control and use of control momentum gyroscopes for the attitude control avoids the necessity of impulsive manoeuvres during the free-flying period, which leads to an improvement of µg-conditions compared to previous permanently manned space stations. On the other side the propulsion system gets more complex and more than 14% heavier. The biggest part of propellant is needed for the rendezvous and docking manoeuvres. They have to be performed with higher thrust levels by chemical propellant. If the requirement to proceed the docking within one work shift (6 hrs) could be relaxed, propellant could be saved.

9. Data Handling System

9.1. Requirements and Assumptions

- 10 years operations, 15 years by replacement
- Man-tended spacecraft requires a two failure tolerant control system to provide
    - Control of hazardous function leading to loss of crew
    - Safe disposal of the spacecraft (ground safety)
- Architecture shall base on commercial available technology (COTS items).
- The intelligent subsystem controllers are connected to this LAN for data exchange with
    - the central spacecraft command and control unit for system supervision and control
    - communication to flight crew
    - inter-orbit communication
    - to/from ground

9.2. Data Volume Requirements

Draft figures for the equipment dimensions are provided in paragraph 9.5.1. to support the accommodation assessment.

9.3. Baseline Design

- As spacecraft internal data exchange back-bone a deterministic local area network has been selected which supports real-time application (close-loop control) and allows the data transfer of asynchronous data. The available technology allows the implementation of a triple-LAN system 1000BaseT.
- The triple LAN system is designed for real-time application and asynchronous services like file transfers and allows on the same physical media the combination of
    - command & control application
    - Medium and high rate data distribution
    - Time distribution
    - Video streaming

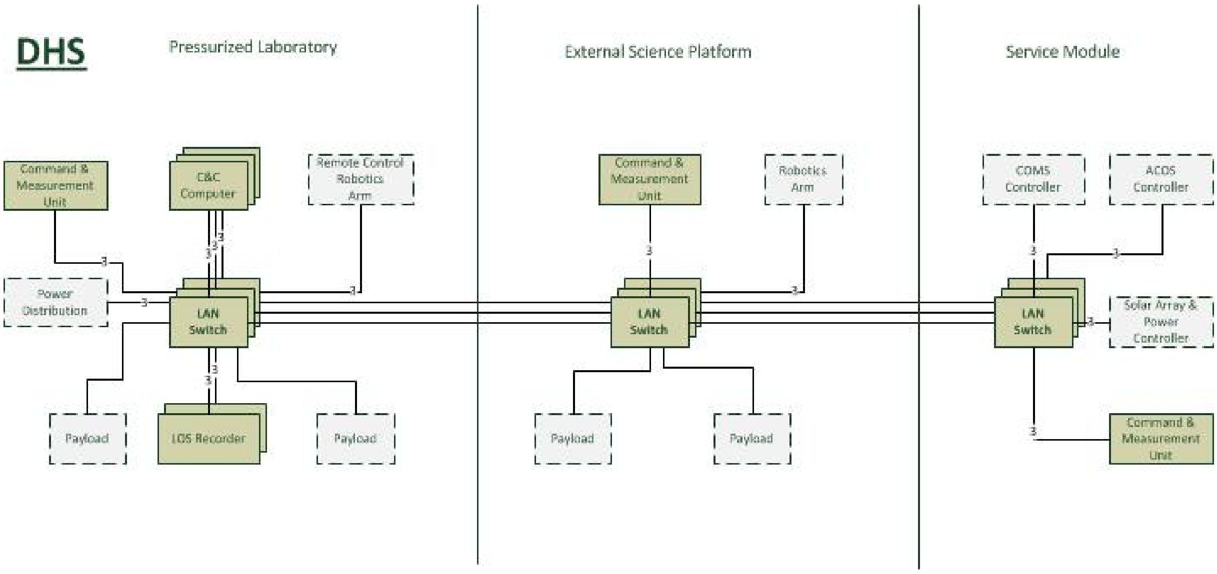


Figure 9-1: Free-Flyer DHS concept.

- A set of Command/Control Units (3 hot redundant computers) are foreseen as central supervision and control instance. The units contain the application data and software storage in non-volatile memory.
- The data handling architecture provides for distributed sensors and actuators the interface conversion between the LAN and the standard hardware, to minimize the modification of existing hardware.
- A LOS data buffering and data compression unit is provided as a central service. For a loss of sight (LOS) duration of up to 30 min (including margin) data storage is provided with the replay priority lower than real-time data. Base on the science requirement of handling up to 2 times 3.3 TByte of data per day (Earth observation, two instruments) a storage volume of minimum 140 GByte is required.

9.4. Options and Trades

- Existing interface standard (1553B MIL Bus, 10BaseT LAN, high rate data using TAXI protocol) vs. Unified data exchange system
- Standard compression vs. Extendable system

9.5. Mass and Power Budget

9.5.1. List of Equipment

- Command & Control Unit [CCU]
  - Dimension: 370x170x200
- LAN Switch [LANS]
  - Dimension: 260x190x70
- Command & Monitoring Unit [CMU]
  - Dimension: 370x320x225
- LOS Recorder [LOSR]
  - Dimension: 470x170x200

Table 9-1: Mass budget of the DHS equipment.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
ESP_Command_Measurement_Unit	15.00	20.00	3.00	18.00	11.78
ESP_LAN_Switch	18.00	5.00	0.90	18.90	12.37
PL_Command_Control_Unit	21.00	10.00	2.10	23.10	15.11
PL_Command_Measurement_Unit	15.00	5.00	0.75	15.75	10.30
PL_LAN_Switch	18.00	5.00	0.90	18.90	12.37
PL_LOS_Recorder	30.00	10.00	3.00	33.00	21.59
SE_Command_Measurement_Unit	15.00	5.00	0.75	15.75	10.30
SE_LAN_Switch	9.00	5.00	0.45	9.45	6.18

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	141.00			152.85	

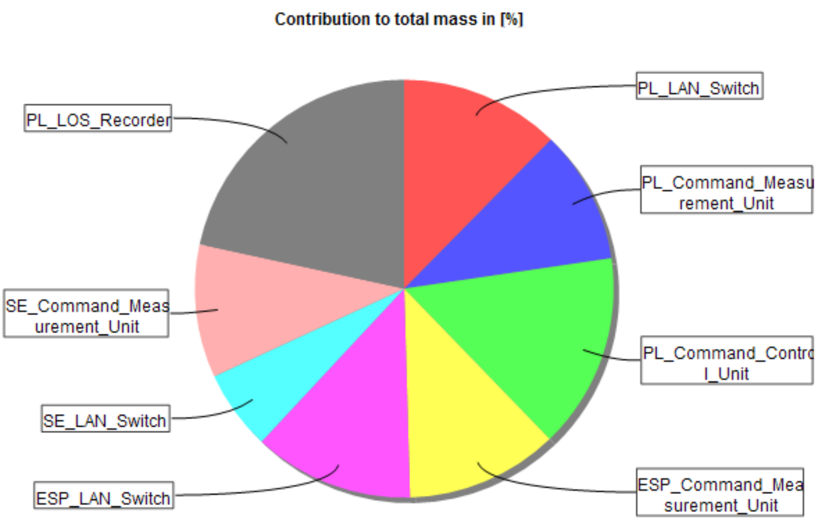


Figure 9-2: Pie chart of mass distribution of the DHS equipment

9.5.2. Power Budget

Table 9-2: Power budget of the OBC system.

System Component	Parameter	Unit	Standard Mode	Survival Mode
DHS	power_avg_wMargin	Watt	575.250	486.000
ESP_Command_Measuremen	power_avg_wMargin	Watt	30.000	0.000
ESP_LAN_Switch	power_avg_wMargin	Watt	157.500	157.500
PL_Command_Control_Unit	power_avg_wMargin	Watt	66.000	66.000
PL_Command_Measurement_	power_avg_wMargin	Watt	26.250	0.000
PL_LAN_Switch	power_avg_wMargin	Watt	157.500	157.500
SE_Command_Measurement_	power_avg_wMargin	Watt	26.250	26.250
SE_LAN_Switch	power_avg_wMargin	Watt	78.750	78.750
PL_LOS_Recorder	power_avg_wMargin	Watt	33.000	0.000
DHS	power_energy_wMargin	Joule	41749344000...	83980800.000

9.5.3. Mode dependencies

The data handling system is used for the data collection and command distribution to provide the status data and the minimum and standard system control. As a consequence of the selected avionics architecture the spacecraft internal data exchange relies on the deterministic local area network, which needs to be active in the survival mode, too. The LOS recorder is only needed for routine operations.

9.6. Re-Supply Items

- none

9.7. To Be Further Studied / Additional Considerations

- Distributed vs. Central control concept
- Safety requirements driving the DHS architecture
- Need of independent supervisor layer (abort function, red button)
- Need for 10GBit backbone
- Design constraints on software architecture
- Use of COTS items in critical functions for man-rated systems
- Commonality with Orbital-Hub on architecture/equipment level
- Automatic mission execution to bring the operations cost down

9.8. Summary / Comparison to ISS

- The ISS provides dedicated infrastructure for system data handling, payload operations, high rate data transfer and video. In contrast the Free Flyer uses an integrated system for data handling and high data/video distribution

10. Communication and Ground Segment

10.1. Requirements and Assumptions

- The communication system will provide TT&C and data communications, both on the platform and to ground
- The communication system should be capable of channelling communications via ground stations and GEO data relay systems

10.2. Ground Station Contacts

No specific contact scenario was defined within the study. A more elaborate communications scheme would be necessary, including the definition of critical or time-sensitive data, before a minimum number of contacts and operation requirements can be set.

10.3. Baseline Design

- External communications include 3 channels:
    - Optical (laser) communications for payload data due to high data volume
    - S-Band: Command, Telemetry, Docking, emergency TT&C and Audio Channels
    - K-Band: Support/back-up for Payload, Video data, others

- Communication system elements:
    - Laser communications equipment
    - Antennas: 1 K-band, 2 S-band
    - Transponders: 1 K-band, 2 S-band

- Internal communications include: WiFi access, and Ethernet (1000base) as a backbone.

<u>*Laser Communications to be channelled through TDRS/EDRS (main datalink):*</u>

- Advantages:
    - High contact time (estimated no less than 60 min/orbit)
    - EDRS will provide optical comms with data rate 1,8 – 7,2 Gbps (6-26 Tb per orbit)

- Disadvantages:
  - Requires high stability (precise pointing to GEO)
  - Rolling motions must be compensated through steering
  - Real time communications suffer delay due to loop through GEO

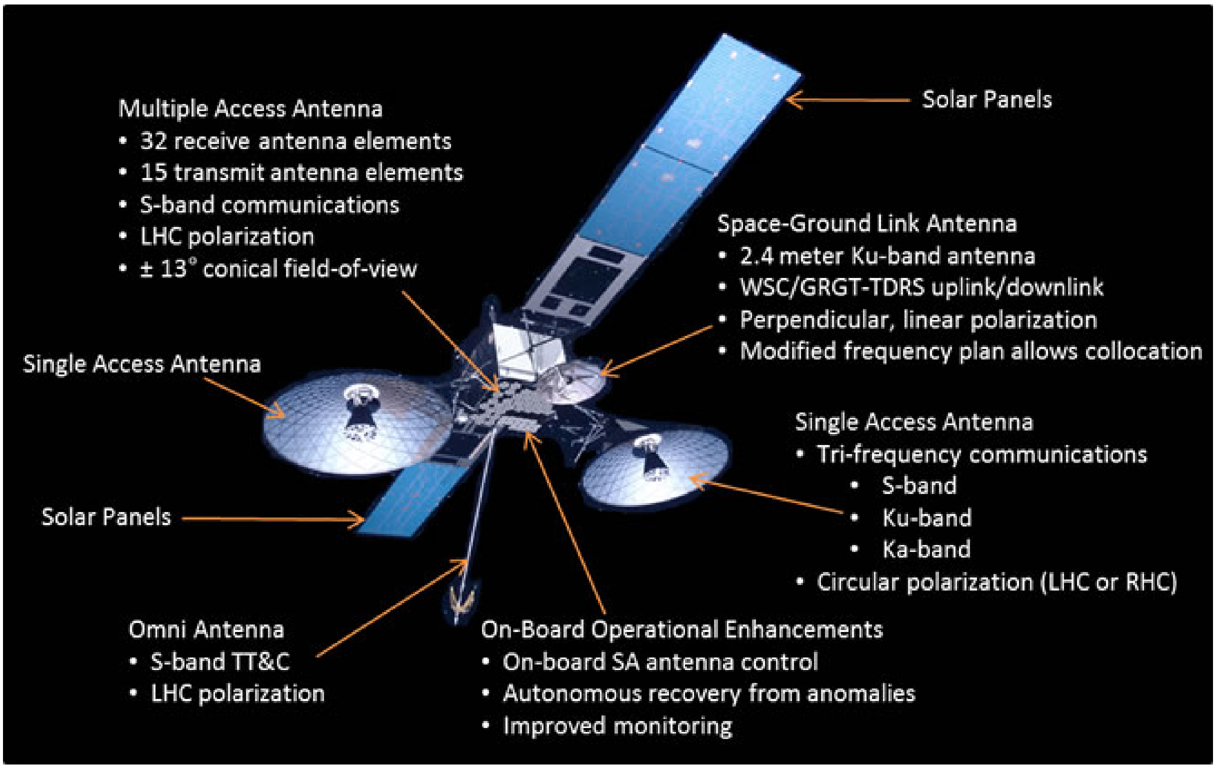


Figure 10-1: NASA TDRS (second generation).

S-band Communications:

- Service Module component: TT&C communications to ground
- Pressurized Laboratory component: Docking communications, and emergency relay link for TT&C

K-band Communications to Ground (back-up system/secondary datalink):

- Current Ku ground capabilities (updated 2013):
  - Uplink: 25 Mbps (pre-2013, 3 Mbps)
  - Downlink: 300 Mbps (pre-2013, 150 Mbps)
- To be used when laser communications not capable of transmitting all scientific data, as a possible back-up to S-band for TT&C, or for additional data transfer

Figure 10-2: ESATRACK (Ground Segment Distribution).

<u>Internal communications:</u>

- Use of Gbit WiFi (access point on the External Service Platform) for payload data transfer
- Gbit Ethernet (1000base) as communications backbone in the Free-Flyer

10.4. Options and Trades

As no specific communication data transmission requirements are yet set, the design defined above can change substantially if significantly less communication capabilities are desired.

This configuration including laser communications would provide a higher communication capability than the one currently used in the ISS, but if the data requirements is low enough, the laser could be eliminated and all communications be undertaken through the S-Band and K-Band channel.

We propose to use the same configuration (systems) in the Base Station, which would potentially simplify implementation (possibly optimise costs), and provide the option of using either module as a data-relay of the other in case of communication system failure.

10.5. Mass and Power Budget

10.5.1. List of Equipment

Table 10-1: Mass budget of the communication equipment.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Ethernet_and_WiFi	40.00	10.00	4.00	44.00	12.21
PL_K_Band_Transponder	6.00	20.00	1.20	7.20	2.00
PL_RF_Distribution_Unit	10.00	10.00	1.00	11.00	3.05
PL_S_Band_Antenna	4.00	10.00	0.40	4.40	1.22
PL_S_Band_Transponder	4.00	10.00	0.40	4.40	1.22
SM_K_Band_Antenna	43.00	5.00	2.15	45.15	12.53
SM_Laser_Comms	200.00	20.00	40.00	240.00	66.60
SM_S_Band_Antenna	4.00	5.00	0.20	4.20	1.17

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	311.00			360.35	

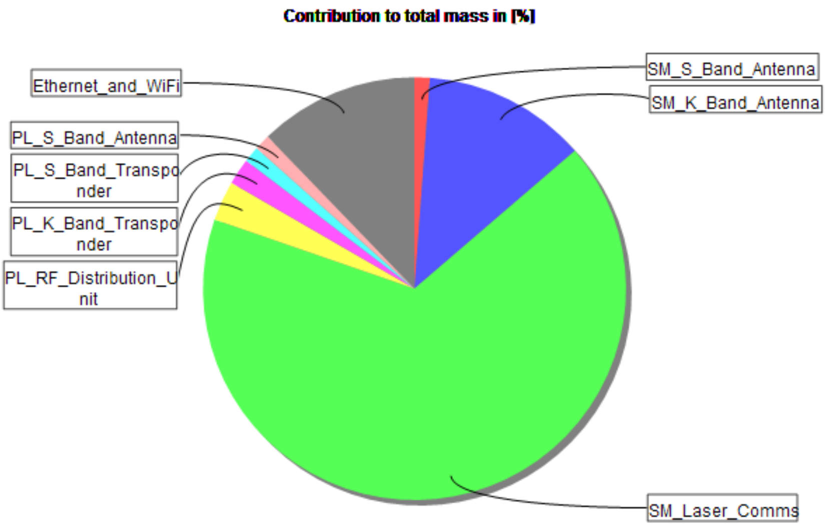


Figure 10-3: Pie chart of mass distribution of the communication equipment

10.5.2. Power Budget

Table 10-2: Power budget of the communication system.

System Component	Parameter	Unit	Standard Mode	Survival Mode
Communication	power_avg_wMargin	Watt	790.800	330.000
SM_S_Band_Antenna	power_avg_wMargin	Watt	0.000	0.000
SM_Laser_Comms	power_avg_wMargin	Watt	295.200	0.000
SM_K_Band_Antenna	power_avg_wMargin	Watt	0.000	0.000
PL_S_Band_Transponder	power_avg_wMargin	Watt	220.000	220.000
PL_S_Band_Antenna	power_avg_wMargin	Watt	0.000	0.000
PL_RF_Distribution_Unit	power_avg_wMargin	Watt	55.000	55.000
PL_K_Band_Transponder	power_avg_wMargin	Watt	165.600	0.000
Ethernet_and_WiFi	power_avg_wMargin	Watt	55.000	55.000
Communication	power_energy_wMargin	Joule	57393100800...	57024000.000

10.5.3. Mode dependencies

The main communication channels (Laser and K-band) wound be unnecessary during survival mode. Internal communications and S-band would need to be available for communication with the crew, as well as for TT&C.

10.5.4. Ground Station Data Handling

Since both the laser and K-band communications are to be channelled through the a geostationary data relay system, the use of ground stations will be needed only for S-band communications in case they cannot be established through the data relay system (e.g. in emergencies).

10.6. Re-Supply Items

- Communication equipment: lifetime 8-15 years

10.7. To Be Further Studied / Additional Considerations

- Communication technology/standards are advancing rapidly; it is difficult to foresee how much higher data transfer technologies (e.g. LiFi, optical ethernet) will have penetrated the market in 10-15 years
- Specific data transfer requirements must be established (payload data), and operational aspects considered
- Laser communications interference by solar array must be further analysed

10.8. Summary / Comparison to ISS

- The communication systems proposed provides much higher data rates than current ISS
- We propose to use the same configuration (systems) in the Base Station, which would potentially simplify implementation (possibly optimize costs), and provide the option of using either module as a data-relay of the other in case of communication system failure.

11. Power

11.1. Requirements and Design Drivers

The stations Power Subsystem shall be designed to handle the following requirements:

- Average power demand of 20 kW
- ISS-like orbit –maximum 36 minutes eclipse duration @ 400 km
- Survival Mode: 1 orbit completely without photovoltaic power
- Station lifetime: 15 years

11.2. Modes of Operation and Design Cases

The station will operate in two different modes;

- Standard,
- Survival
    - Standard Mode (normal operation before incident)
    - 36 minutes max. Eclipse → total loss of PV power for 1 orbit between 2 eclipses.

The Standard mode corresponds to the Free-Flyers power consumption during normal operation. This mode will be the most critical and will therefore be used to size the battery and the solar panels. The baseline demand is for the power system to be able to supply 20 kW to all payloads and subsystems.

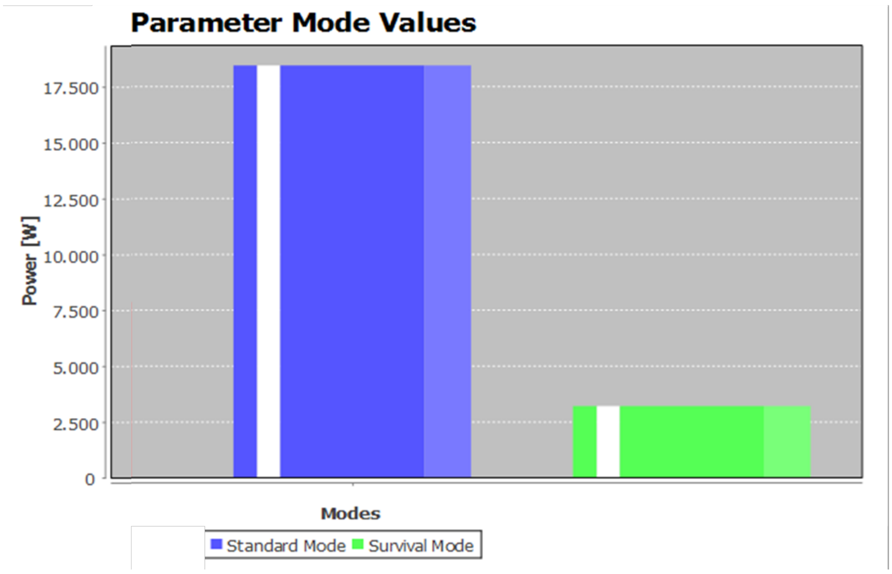
The Survival mode covers the Free-Flyers power consumption after a fail in attitude pointing, when the solar panels are turned away from the sun. In this mode, the battery needs to supply the station for one and a half orbit. This is an easily met requirement considering the low power demand during this mode.

11.3. Power Budget

The table below (Table 11-1) shows the power requirements for the Free-Flyers subsystems during the two operation modes. The values marked with orange are the powers required downstream from the battery, which is subject to photovoltaic efficiency, charging cycle efficiency, and power conversion efficiencies equal a system margin of 20%. This also includes Power Subsystem internal regulated power required for control equipment and communication with the on-board data handling. The values marked with red are the powers required by the Payloads and Subsystems.

Table 11-1: Overall power budget of the Free Flyer.

System Component	Parameter	Unit	Standard Mode	Survival Mode
FreeFlyer	mode_duration	Second	72576000.000	172800.000
FreeFlyer	power_avg_wMargin	Watt	15352.460	2647.459
FreeFlyer	power_avg_wMargin_wS...	Watt	18422.952	3176.950
FreeFlyer	power_avg_wMargin	Watt	15352.460	2647.459
AOCS	power_avg_wMargin	Watt	423.880	21.000
Communication	power_avg_wMargin	Watt	790.800	330.000
Crew_Workstation	power_avg_wMargin	Watt	0.000	0.000
ECLSS	power_avg_wMargin	Watt	351.321	0.000
DHS	power_avg_wMargin	Watt	575.250	486.000
Propulsion	power_avg_wMargin	Watt	2394.551	3.877
Robotic_Automation	power_avg_wMargin	Watt	60.200	0.000
Structure_Debris_Rad_Prot	power_avg_wMargin	Watt	13.080	13.080
Thermal	power_avg_wMargin	Watt	672.600	1793.500
Venting_Systems	power_avg_wMargin	Watt	1.578	0.002
Harness	power_avg_wMargin	Watt	0.000	0.000
Payload_Science	power_avg_wMargin	Watt	10069.200	0.000
FreeFlyer	margin_system	Percent	20.000	20.000
FreeFlyer	power_energy_wMargin	Joule	17800323711...	545297819.040
FreeFlyer	power_energy_wMargin_...	Joule	21360388453...	654357382.848



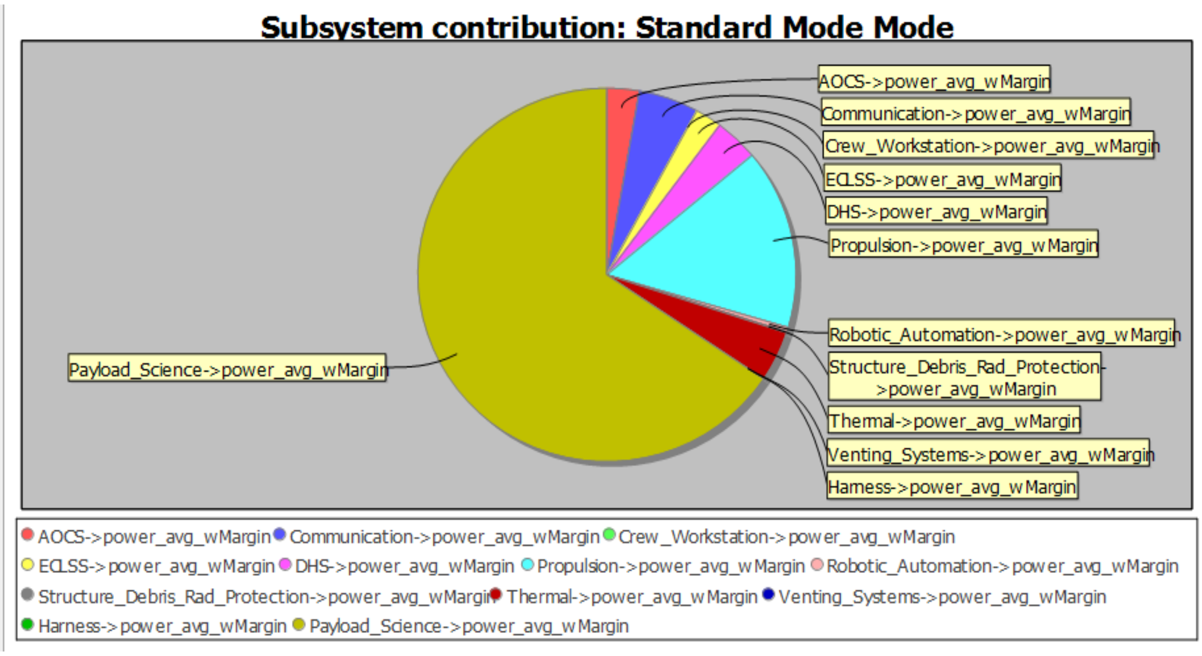


Figure 11-1: Overall power budget of the Free Flyer.

11.4. Baseline Design

The baseline design for the power subsystem is the same as for the Base Station and is based on the current ISS design but with increased efficiency and simplicity. The design of the power subsystem will assure redundancy using the following configurations:

- 3 times redundancy on the primary bus
- No redundancy, but double connections on the secondary bus
- DC/DC conversion done at P/L and S/S level to increase the system efficiency
- Massively parallel design on power generating system to increase the reliability.

The power sub system on the ISS uses a lot of steps for voltage down-conversion while this new design uses only two steps, one between photovoltaics and the battery for controlled charging, and one at payload or bus unit level. The voltage down-conversion is an essential part of the system since the relatively high battery bus voltage needs to be converted to much lower voltages required by electronics and small actuators. Thus, the harness is operated at the highest voltage widely used in the power subsystem and its resistance contributes as little as possible to losses by voltage drop.

This new design is possible since the Li-ion battery cell chemistry proposed does not require any dedicated maintenance and conditioning operations, as the Ni-based battery cells originally used on the ISS. In fact, battery handling is virtually carefree as long as the minimum voltage, maximum voltage, and maximum current limits are observed.

However, managing the average state of charge can significantly reduce battery ageing (reduction of maximum capacity) and thus increase useful battery lifetime [RD 10].

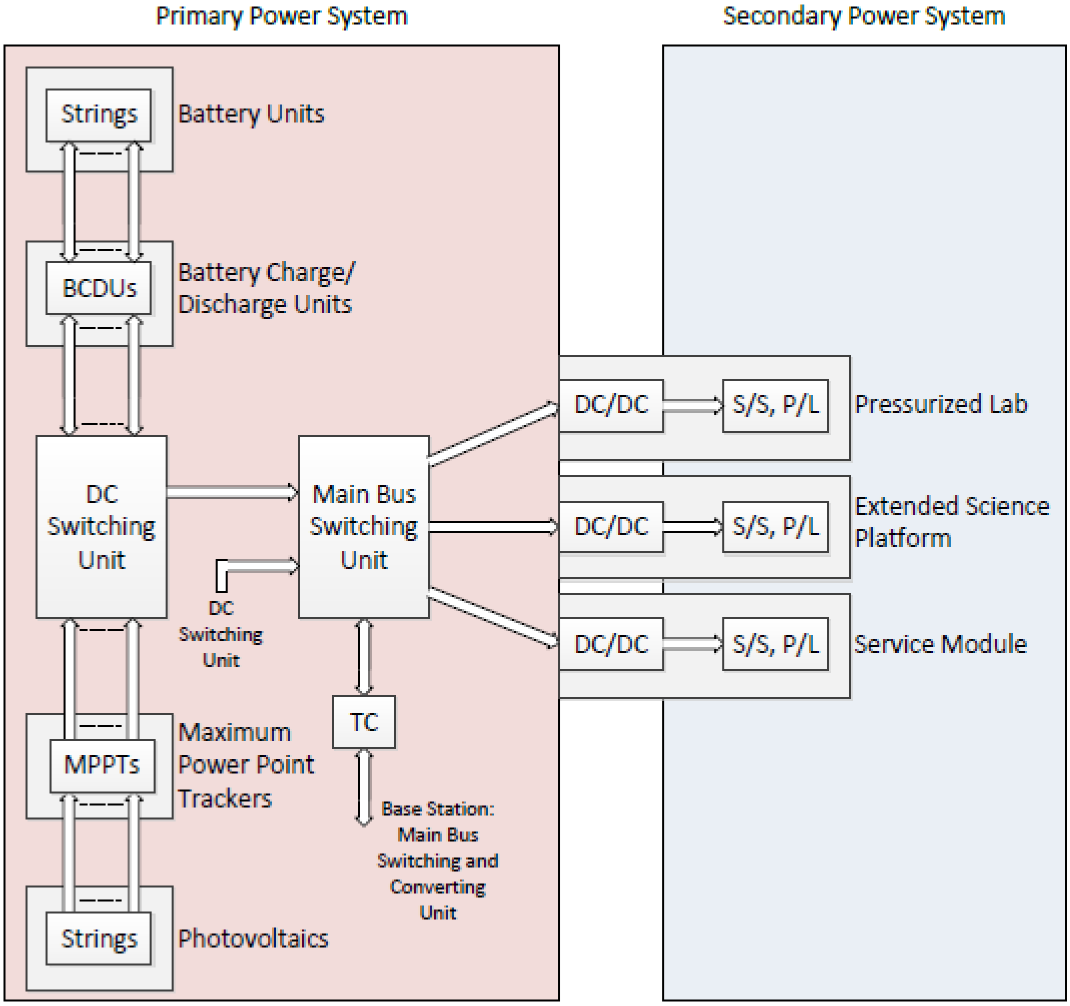


Figure 11-2: Power subsystem architecture.

The designs of the Power subsystem can be seen in the figure above (Figure 11-2). The design consists of solar panels that are connected to a DC Switching Unit (DCSU) via Maximum Power Point Trackers (MPPT). The DCSU is a matrix of protected circuit breakers that connects different branches of photovoltaics, battery modules, and power buses. Also connected to the DCSU are the batteries and the Battery Charge/Discharge Units (BCDU) which in this design are not power converters but mainly switching units to manage (engage, disengage in a controlled manner) sub-units of the battery which is very large compared to other LEO spacecraft. (A byway power converter may be included to re-equalize the state of charge of a battery module for re-connection; however this is no operational power path.) Every module of the battery will be connected to its own BCDU to get more reliable system.

The power is routed by the DCSU and sent to a Main Bus Switching Unit (MBSU). The MBSU will distribute the power to all payloads and bus units that require it and of whom the majority will also do the second and final voltage conversion step themselves. For some power users requiring a common voltage at significant power level there may be centralized power conversion already in the MBSU and/or the MBSU may act as the controlled power switch output for other bus units towards a high-power actuator.

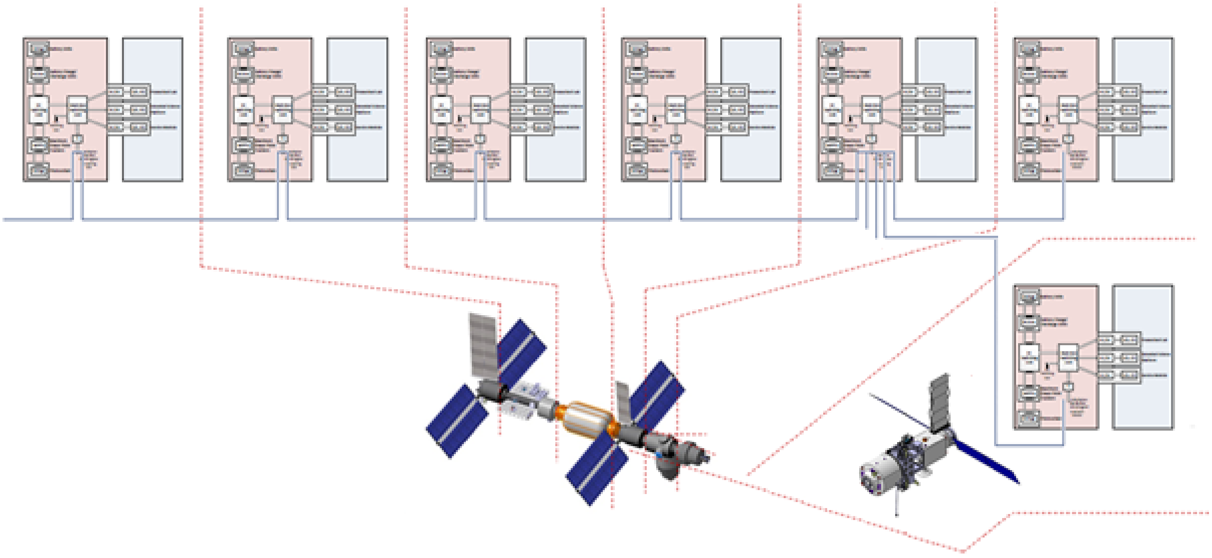


Figure 11-3: System topology of the power subsystem with multiple modules.

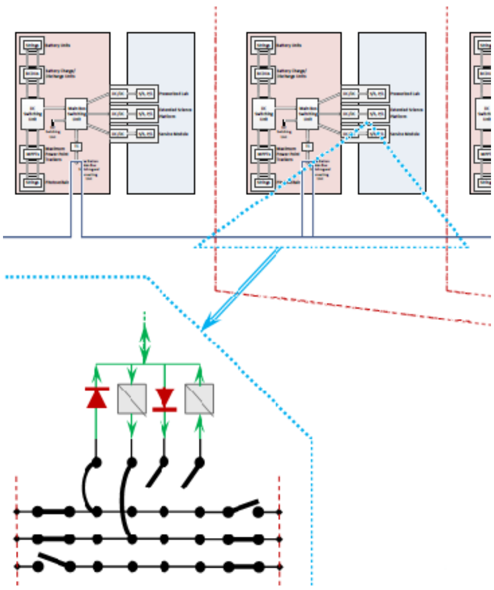


Figure 11-4: Detail of power transfer architecture concept.

The MBSU can also be connected to other MBSUs, via the Transfer Converter (TC). The TC enables a link between the different MBSUs on the complete station, thus allowing the Free Flyer to easily dock and undock. An advantage of this concept is that only the amount of power that 'spills over' from section to section of the station has to undergo power conversion with a significant loss factor. If the docking/berthing interface of the station modules enables this, several power buses can run down the length of the station and into its branches, with each module's TC feeding to and/or drawing from any configuration of these buses through a set of selection switches. A representation of this can be seen in Figure 11-4 and Figure 11-6 .

11.5. Mass and Power Budget

11.5.1. List of Equipment

Table 11-2: Mass budget of the power equipment.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
EX_DCDC	120.00	10.00	12.00	132.00	5.30
PL_DCDC	120.00	10.00	12.00	132.00	5.30
SE_Battery	956.34	10.00	95.63	1051.97	42.26
SE_BCDU	200.00	10.00	20.00	220.00	8.84
SE_DCDC	120.00	10.00	12.00	132.00	5.30
SE_DCSU	200.00	10.00	20.00	220.00	8.84
SE_MBSU	100.00	10.00	10.00	110.00	4.42
SE_MPPT	87.00	10.00	8.70	95.70	3.84
SE_SolarPanel_Electronics	50.00	10.00	5.00	55.00	2.21
SE_SolarPanels_PV	175.00	5.00	8.75	183.75	7.38
SE_SolarPanels_Structure	142.80	10.00	14.28	157.08	6.31

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	2271.14			2489.50	

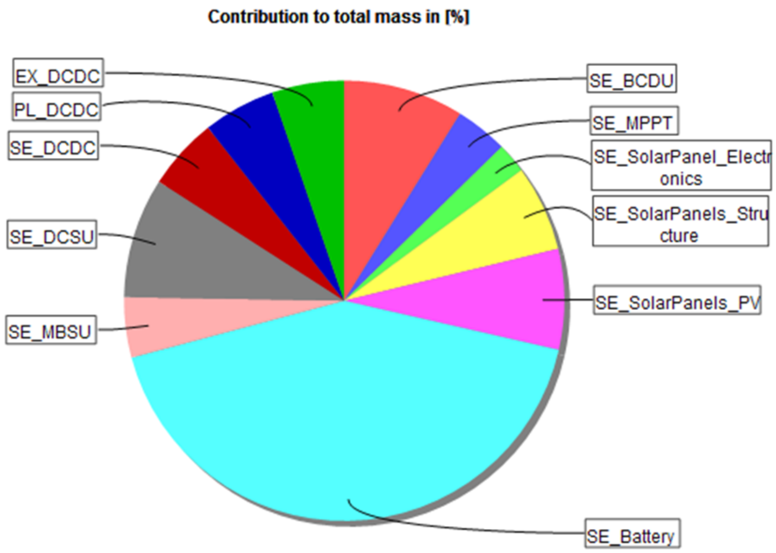


Figure 11-5: Pie chart of mass distribution of the power equipment

11.6. Options and Trades

During the post-processing of the CE-study the study assumptions have been revised and are described in more detail in the following sections:

The primary system design will be focused on a massively parallel topology, which will provide a very stable and fault tolerant system. This design is illustrated in the figure below. The design will be focused on the following aspects:

- panel level – 4 to 16 blocks corresponding to mechanical panel structure segments
  - largely traditional design with modular redundancy
- kW / kWh level – 10's of parallel blocks with ~1 kW photovoltaic power
- single PV string – 100's of parallel blocks with ~50 W photovoltaic power
  - integrate DCSU switch in each MPPT-BCR
  - graceful degradation

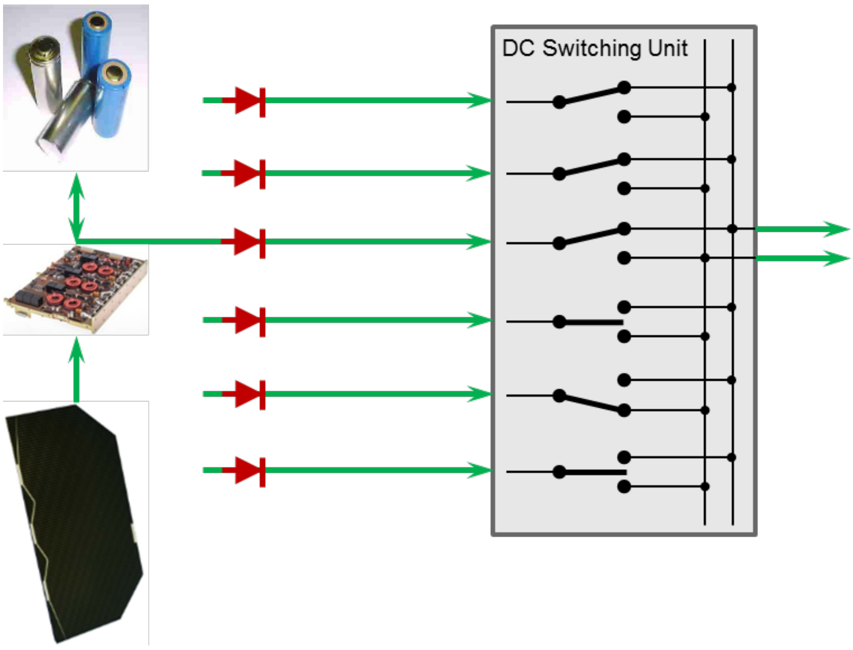


Figure 11-6: System Topology option of the power subsystem (parallel design)

11.6.1. Power System Losses

The losses in the power system have been estimated to the values seen in the figure below. The losses will vary depending on if the secondary power system is supplied directly from the solar panels during the sun phase or via the battery during eclipse. The different power losses in these two cases are:

$$Photovoltaic\ power\ losses\ ECLIPSE : P_{PV} \cdot 0.95^3 \cdot 0.8 = 0.6859 \cdot P_{PV}$$
$$Photovoltaic\ power\ losses\ SUN : P_{PV} \cdot 0.95 \cdot 0.8 = 0.76 \cdot P_{PV}$$

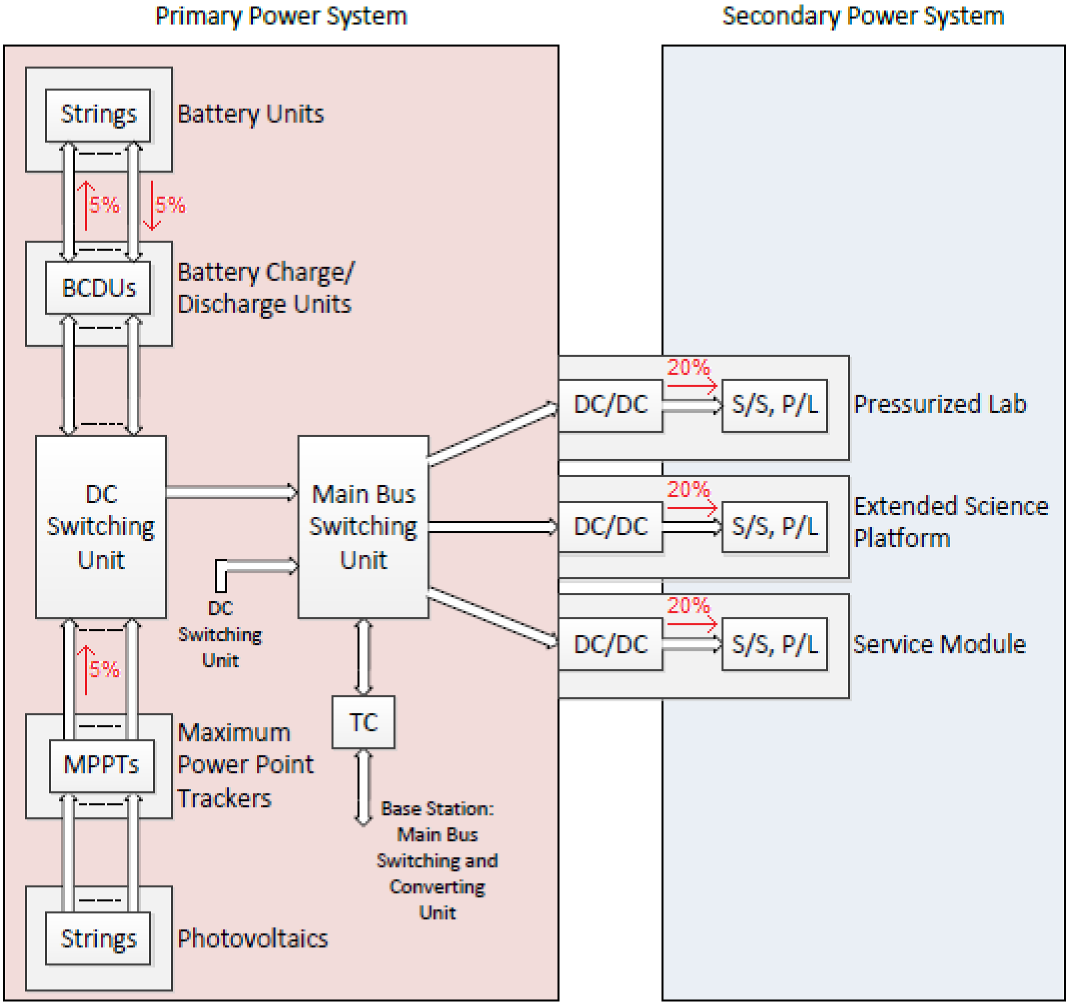


Figure 11-7: Power system losses.

The voltage at the DC/DC Converter Units (DDCU) inputs on ISS varies between 133-177V, this minimum value was also selected for the Free Flyers power system. A minimum voltage of 133 V at the DDCUs would require a minimum photovoltaic and battery output voltage of:

$$V_{PV}(min) = V_{Battery}(min) = \frac{133}{0.95} = 140\ [V].$$

11.6.2. Systems Design

Some of the units used in the proposed system design are not available and have to be specially made for the stations unique requirements. The units that do not require this have been designed while the other ones have been scaled with the corresponding unit on the ISS.

Solar panels

The selected solar cells are the Triple-Junction GaAs solar cells from AZUR SPACE. The cells have the following electrical characteristics:

Figure 11-8: Solar cell.

- Cover glass and monolithic integrated bypass diode
- Worst case current (EOL 70°C): 0.49836 A
- Worst case voltage (EOL 70°C): 1.9436 V
- Mass per cell: 0.00356124 kg
- Area per cell: 0.003018 m²

The amount of power that needs to be generated during sunlight to supply the power requirements during eclipse is calculated using the power requirement of 20 kW, the power losses and the time duration for sun and eclipse:

$$P_{PV} = \left(\frac{20000 \cdot 55}{0.76} + \frac{20000 \cdot 36}{0.6859}\right) \cdot \frac{1}{55} = 45402\ [W]$$

The power requirement can then be used to size the total solar panel configuration in serial and parallel strings:

$$Cells_{Serial} = \frac{140}{1.9416} = 73\ cells$$
$$Cells_{Parallel} = \frac{45402}{73 \cdot 1.9416 \cdot 0.49686} = 645\ cells$$

The total amount of cells is then equal to:

$$Quantity_{PV\ cells} : 73 \cdot 645 = 47085\ pcs$$

Other parameters that need to be taken into account when estimating the total mass of the solar panels are specified in the table below. It is assumed that the Free Flyer will have two solar panels, each having a length of 22 m. The complete configuration for both solar panels will have a total mass of 460 kg. The Definitions as a percentage of the photovoltaic blanket mass has been scaled after the ISS. [RD 12]

Table 11-3: Solar panel mass estimation.

Unit Name	*Mass [kg]*	*Definition*
Photovoltaic blanket mass ($Mass_{PV}$)	167.6810	47085·0.00356124
Miscellaneous integration	114.8615	$Mass_{PV}$·0.685
Electrical equipment	114.8615	$Mass_{PV}$·0.685
Mast mass	62.2097	$Mass_{PV}$·0.371
Total Mass	**459.6137**	

The total area, using a cell spacing of 15%, is estimated to 167.2 m² according to:

$$Area_{PV} = \frac{47085 \cdot 0.003018}{0.85} = 167.18\, m^2$$

Battery

The selected battery cells are the lithium-ion 18650HC cells from ABSL. The cells have the following electrical characteristics:

- Capacity of 130 Wh/kg
- Minimum cell voltage of 2.5 V
- Operating temperature between -30°C and 60°C
- 2 cell-level safety devices; hard-short-safe up to 8s string, used by NASA JSC/GSFC on EAPU [RD 8][RD 9]

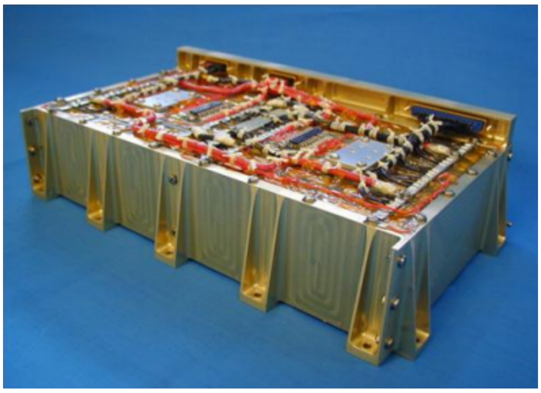


Figure 11-9: Battery stack.

The battery is designed to have a lifetime of 15 years and through this time be able to deliver sufficient power during the flowing modes:

- Standard mode including margin:

$$\frac{20000}{0.8 \cdot 0.95} = 26316W$$

- Survival mode including margin:

$$\frac{2647.459}{0.8 \cdot 0.95} = 3483.5$$

A long mission lifetime puts though constraints on the battery that needs to be operated during optimal conditions to provide sufficient power at the end of the mission. This is achieved by selecting suitable end of charge voltage and depth of discharge values. For the selected cell, different conditions have been tested by the supplier.

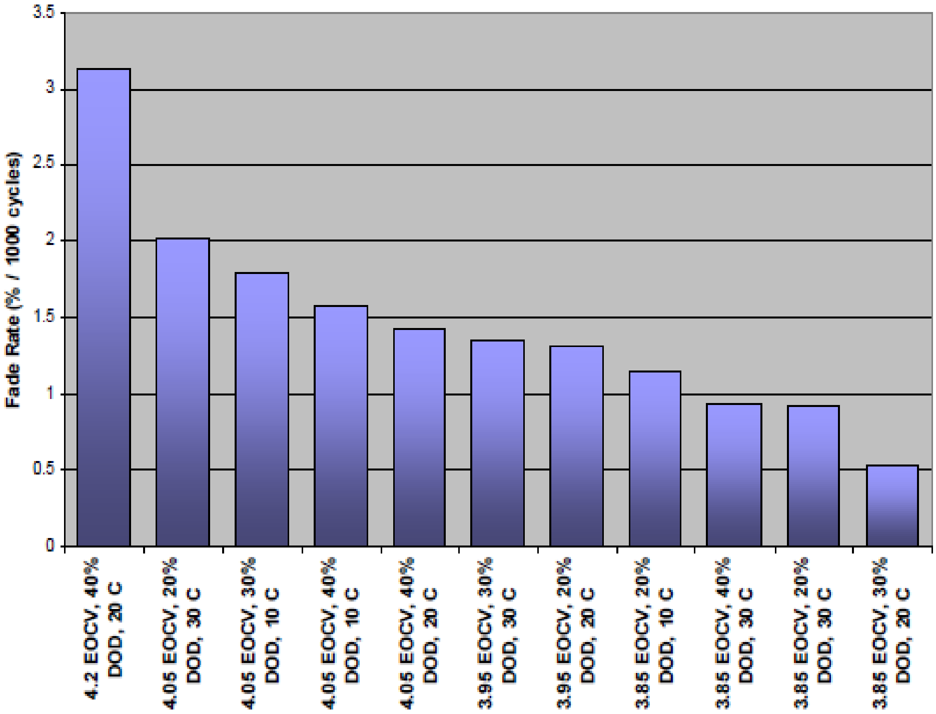


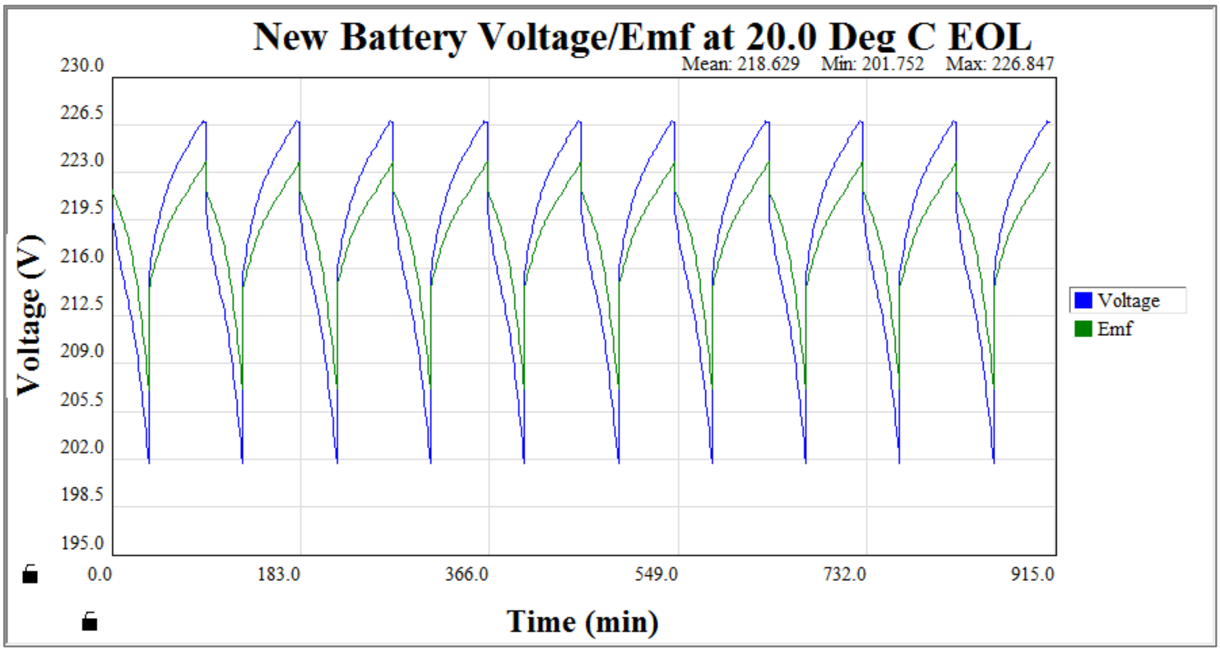
Figure 11-10: Retrievable capacity in the battery depending on discharge conditions [RD 10].

The battery will experience about 86600 cycles which will heavily decrease the capacity, while the huge amount of power requires a high depth of discharge to keep the battery as small as possible. According to these constraints the two most suitable cases from the diagram above was chosen (see Table 11-4).

Table 11-4: Battery simulation cases.

Case	***End of Charge Voltage [V]***	***Depth of Discharge [%]***	***Temperature [°C]***	***Fade Rate [%]***	***End of Life Capacity [%]***
1	4.05 (226.8 total)	40	20	1.4	29.5
2	3.95 (221.2 total)	30	30	1.3	32.3

The two cases were simulated using ABSLs battery simulation tool BEAST. Simulations were first made for the standard mode, confirming that the battery could reach a steady state for the charge and discharge level, also at the end of the mission. Simulations were also made for the safe mode to confirm that the battery would manage during a temporary solar panel pointing failure, and after that recover to the steady state.



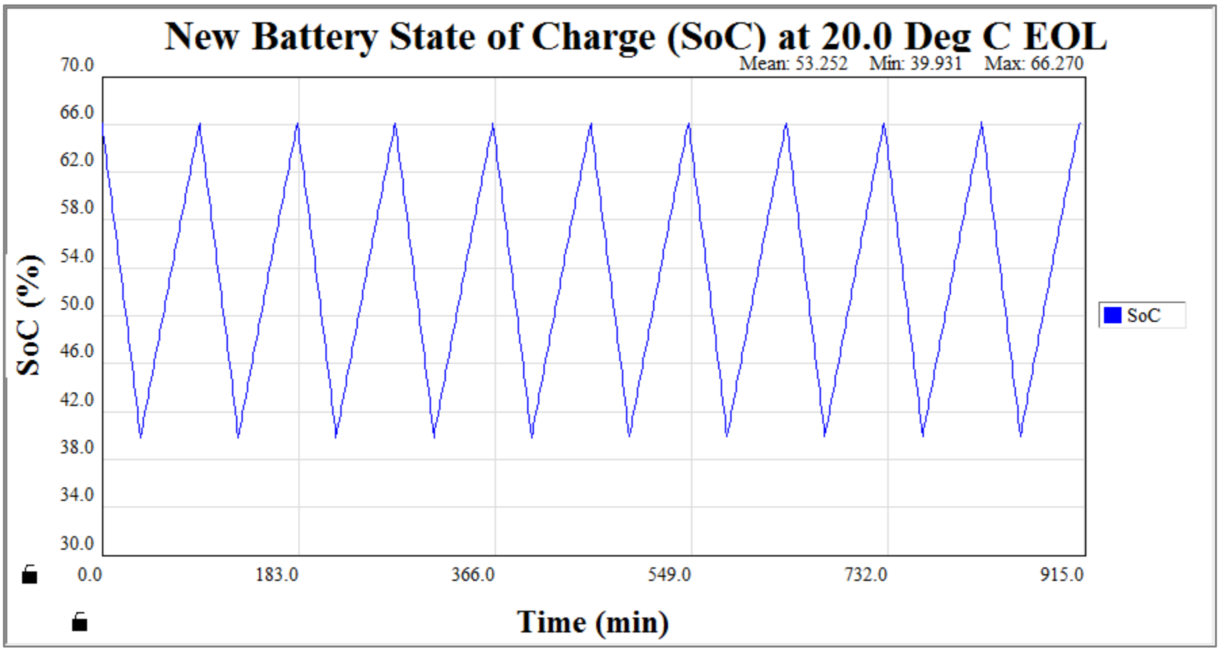
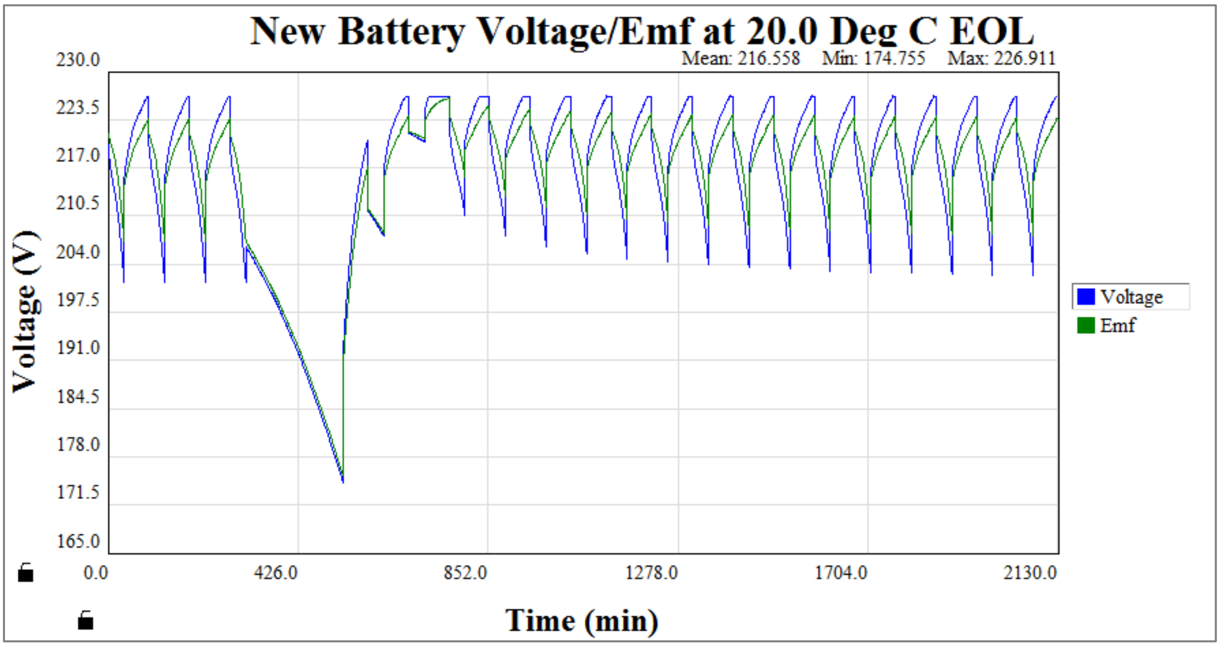


Figure 11-11: Battery state of charge simulation – case 1 – Standard Mode.



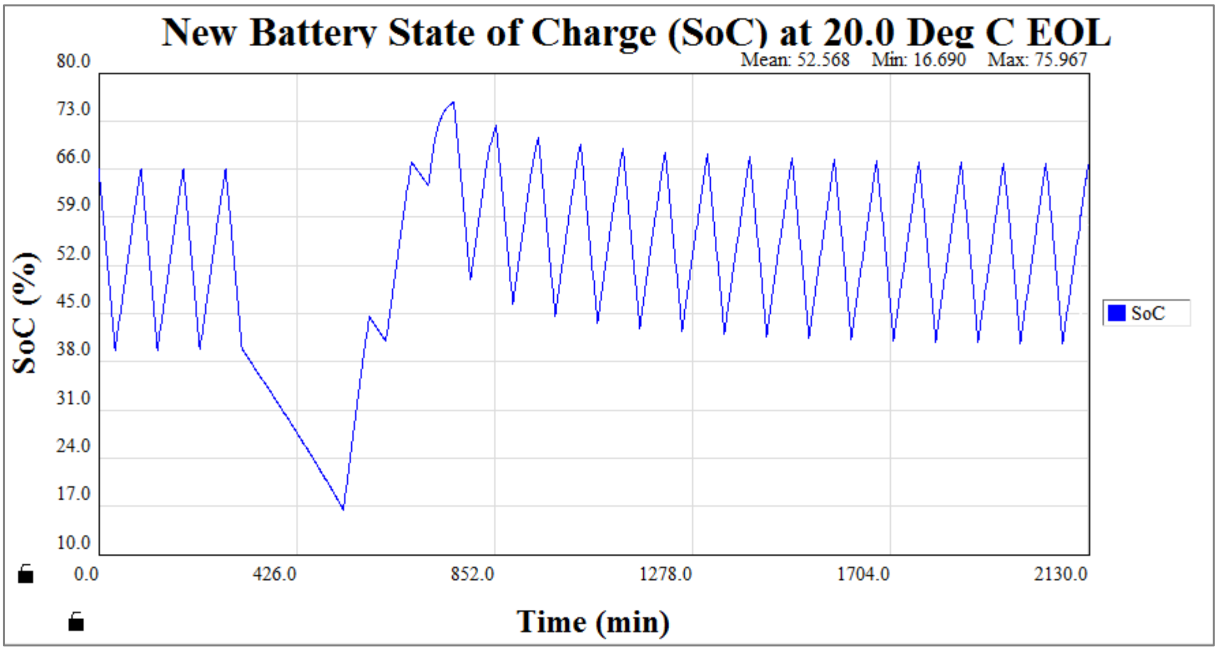


Figure 11-12: Battery state of charge simulation – case 1 – Safe Mode.

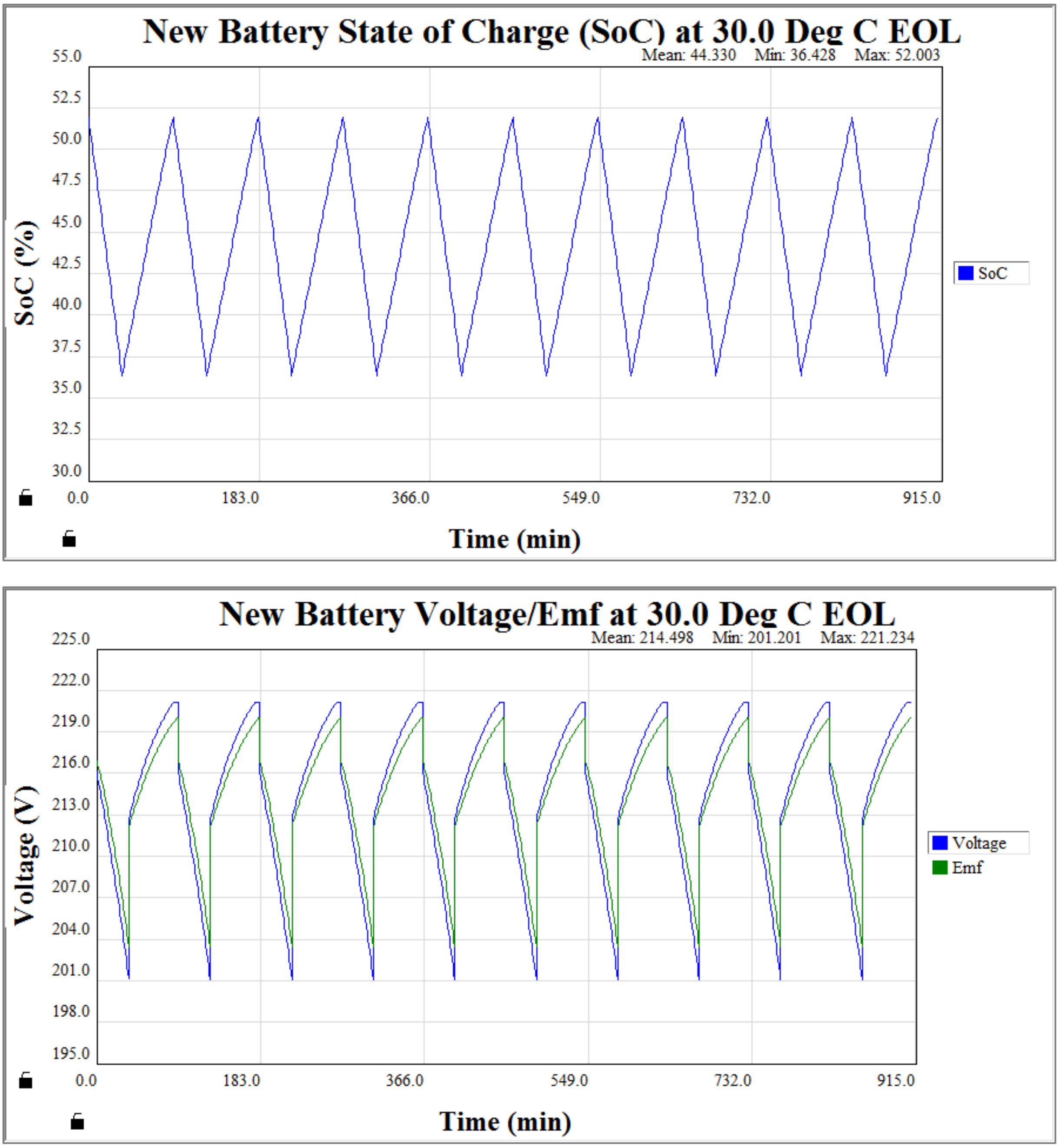


Figure 11-13: Battery state of charge simulation – case 2 – Standard Mode.

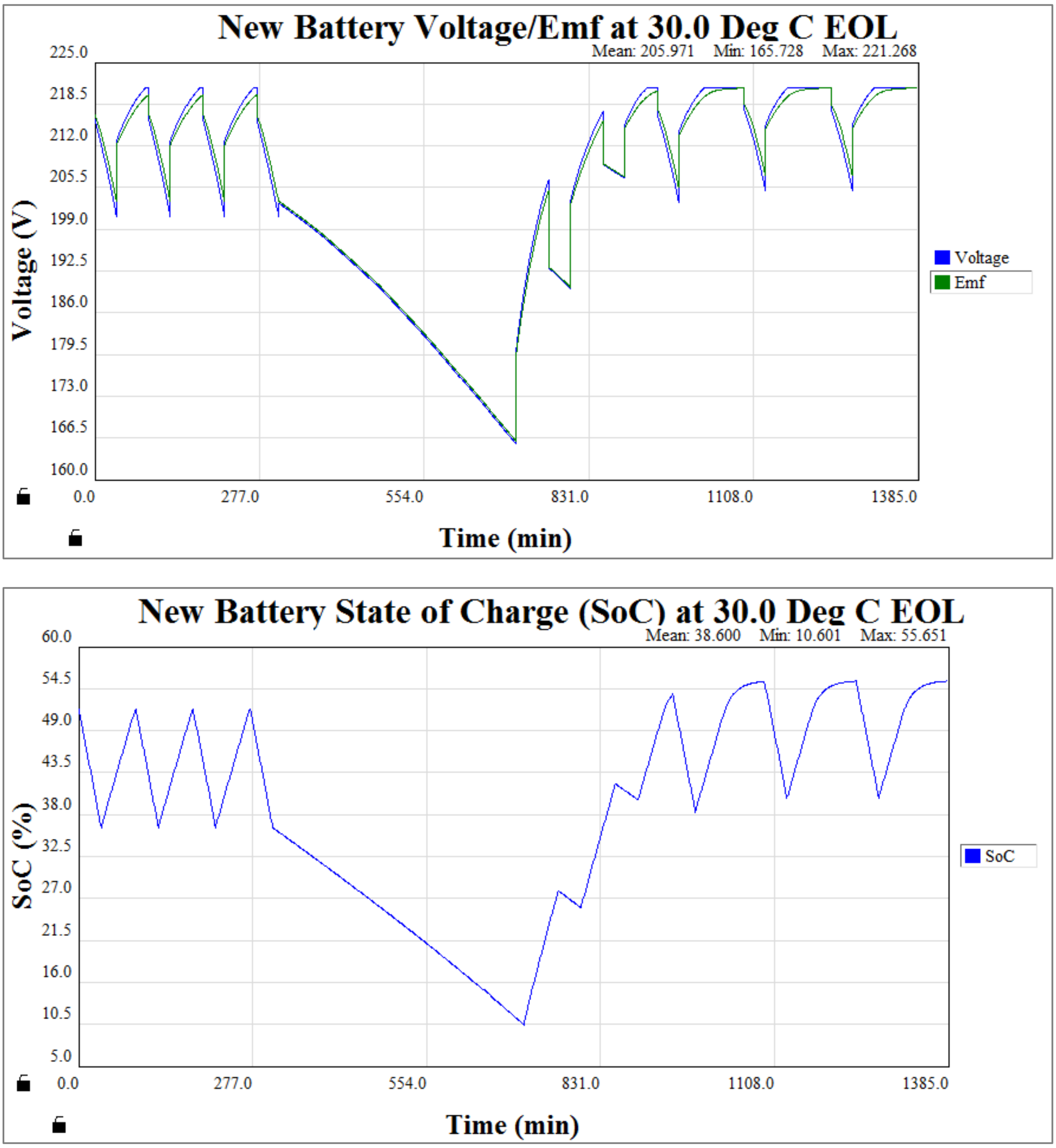


Figure 11-14: Battery state of charge simulation – case 2 – Safe Mode.

Simulation Results

The simulations results can be seen in Table 11-5. They clearly show that case 1 is the most suitable option since it has a much lower mass and still sufficient survival time. Case 1 is therefore chosen as the baseline battery design.

Table 11-5: Battery simulation results.

Case	*18650HC Topology*	*Mass [kg]*	*Survival time without power*
1	56s640p	1682.92	182 min (2 orbits)
2	56s1000p	2629.57	364 min (4 orbits)

Alternative Battery

An alternative battery cell with the same mechanical shape and similar mass per cell could be the ABSL 18650NL, which has the capacity of 190 Wh/kg. Thus, the battery would become much more efficient per mass and volume than the selected cells, but these cells are relatively new and therefore only limited lifetime test information is available as it does not yet have the nearly 15 years of space operational history as the 'HC type.

Electronic Box

The electronic box will contain all the power equipment that will be needed on the Free Flyer. All components needed in this box, except the battery, have been scaled to their equivalent unit on the ISS [RD 11]. The primary system, which includes everything except the DC/DC converters, will be placed in the service module while all three modules will have their own DC/DC converter enabling the power conversion at P/L and S/S level.

Table 11-6: Electronic box mass and volume estimation.

Unit	*No. of units*	*Mass per unit*	*Volume per unit*	*Total mass*	*Total volume*
Maximum Power Point Tracker (MPPT)	2	49.5	0.0765	99	0.150
DC Switching Unit (DCSU)	2	63.5	0.1300	127	0.260
Main Bus Switching Unit (MBSU)	1	118.0	0.2600	118	0.260
Battery Charge/Discharge Units (BCDU)	2	160.5	0.3300	321	0.660
DC/DC Converting Unit (DDCU)	3	25.0	0.0400	75	0.120
Battery	10	168.3	0.1665	1683	1.665
Total				**2198 (+75)**	**2.755 (+0.12)**

The total mass for the primary system including margin is estimated to:

$$Total\ mass\ with\ 10\%\ margin\colon\ M = \frac{2198}{0.9} = 2442.2kg$$

$$Total\ volume\ with\ 20\%\ margin\colon\ V = \frac{2.755}{0.8} = 3.4m^3.$$

The mass and volume for the secondary system is estimated to 75 kg and 0.12 m^3 respectively.

11.7. Re-Supply Items

- No units have to be replaced since they are all designed to have a lifetime of 15 years.

11.8. To Be Further Studied / Additional Considerations

The proposed design needs to be further studied, especially focusing on the further developments from the ISS design. The major things that need to be studied are:

- The new configuration using less converters in the power flow than on ISS
- The cross-feeding between modules, topology optimization for various docking configurations of station modules & possible growth
- Massively parallel PV – MPPT – Battery design (partially assumed)
- A more accurate estimation of the mass and volume for the Electronics box.

11.9. Summary / Comparison to ISS

The power requirements for the Post-ISS Scenario-II study have been fulfilled. This was accomplished using an ISS inspired Power system, but modifying it to a simpler but still as safe and robust system. The new design will also make it easy for the station to develop and grow over the years.

One major design change in the design was the battery. A lot of progress has been made in this area since the ISS was built and todays batteries are much more stable and do not require as much power control for stabile operation. The removal of the many voltage conversion steps will save a lot of power that otherwise would have been lost as heat.

Another improvement was the solar panels that with today's technology have a higher efficiency and does not require the same area as on the ISS. The units required in addition to the battery and solar panels could probably also be smaller than on the ISS, shrinking the mass and volume of the electronic box further, and should be investigated in the future.

12. Thermal

12.1. Requirements and Design Drivers

Heat rejection requirement of Free-Flyer is driven by the system requirement ST-120. Although ST-120 says that Free Flyer shall provide 20kW for housekeeping and payload, 20 kW does not cover any power system's power consumption in the final design (Free Flyer provides 27.44 kW average for the entire system when the maximum power is required). For the thermal system design, heat rejection for the entire Free Flyer is required.

Regarding the operation modes, the standard mode and the survival mode are considered. Under the standard mode condition, 20 kW plus power consumption of the power system is assumed to be the maximum heat load for the thermal system, and power consumption without payload operation is assumed to be the minimum heat load for the thermal system. This power consumption variation is necessary to be handled by standard operation of the thermal system. Under the survival mode condition, the power consumption will be decreased to a least necessary level to maintain Free Flyer. This operation decreases temperature level of Free Flyer, and the thermal system has to prevent each component from over cooling.

In this study, radiation heat exchange within Free-Flyer, between Free-Flyer and Base station, between Free-Flyer and the Earth, and Albedo effect are not considered.

In principle, it is assumed that all power consumption of each component is finally rejected to the space as heat. However, ion engine system is regarded as an exception. The part of the supplied energy to the ion engine is used to accelerate the ionized particles and directly injected to the space. The energy efficiency is estimated based on the following equation using the ion engine performance parameters.

$$Efficiency = \frac{0.5 \times \dot{m} \times V^2}{P} = \frac{0.5 \times FgI_{sp}}{P}$$

$$P: Nominal\ Power\ [W]$$
$$F: Nominal\ Thrust\ [N]$$
$$V: Flow\ velocity\ [m/s]$$
$$\dot{m}: Flow\ rate\ [kg/s]$$
$$g: gravitational\ acceleration\ constant, 9.8\ [m/s^2]$$
$$I_{sp}: Specific\ impulse\ [s]$$

Table 12-1: Ion Engine Performance [RD 13]

	RIT 10 EVO			RIT 2X			
Nominal Thrust [mN]	5	15	25	80	115	168	200
Nominal Power [W]	145	435	760	2165	2985	4650	5785
Isp [s]	1900	3000	3200	3400	3434	4000	4300

For the selected thruster type: RIT 10 EVO with 15 mN thrust and RIT 2X with 200 mN thrust, the estimated efficiency is 50% and 73% respectively. Therefore, the rest of the power, 50% and 27%, is assumed to be dissipated as heat, and ejected to the thermal system. Based on the collected information from every subsystem as shown in Table 12-2, the total power consumption excluding the payload and the power system for standard mode is 3.663 kW, which is much smaller than the maximum power requirement. Since the science payloads are not always under operation, the total power consumption excluding the power system can vary between 20 kW and 3.663 kW in the standard mode operation.

Table 12-2: Summary of Free Flyer Power Consumption

Name	Module	Subsystem	power avg standard [W]	power avg survival [W]
PL_Rendezvous_Docking_Sensor	PL	AOCS	0.043	0.000
ESP_CMG	ESP	AOCS	163.200	0.000
PL_Sun_Sensor	PL	AOCS	0.525	0.000
SE_Horizon_Sensor	SE	AOCS	0.039	0.000
PL_Horizon_Sensor	PL	AOCS	0.039	0.000
SE_Sun_Sensor	SE	AOCS	0.525	0.000
SE_Star_Tracker	SE	AOCS	10.500	10.500
ESP_Rate_Gyro_Assembly	ESP	AOCS	12.600	0.000
SE_GNSS_Receiver	SE	AOCS	16.500	0.000
PL_GNSS_Receiver	PL	AOCS	16.500	0.000
SE_GNSS_Receiver_Antenna	SE	AOCS	1.200	0.000
PL_GNSS_Receiver_Antenna	PL	AOCS	1.200	0.000
PL_Rendezvous_IMU	PL	AOCS	25.200	0.000
PL_Rendezvous_Visual_Camera	PL	AOCS	2.101	0.000
PL_Rendezvous_Visual_Camera_Light	PL	AOCS	0.007	0.000
SE_CMG	SE	AOCS	81.600	0.000
SM_Laser_Comms	SE	Comm	295.200	0.000
PL_RF_Distribution_Unit	PL	Comm	55.000	55.000
PL_K_Band_Transponder	PL	Comm	165.600	0.000
PL_S_Band_Transponder	PL	Comm	220.000	220.000
Ethernet_and_WiFi	PL	Comm	55.000	55.000
Fire_Detection_Suppression	PL	ECLSS	16.500	0.000
Lighting	PL	ECLSS	19.800	0.000
Temperature_Control	PL	ECLSS	150.000	0.000
Ventilation	PL	ECLSS	165.000	0.000

Positive_Pressure_Relieve_Valve	PL	ECLSS	0.021	0.000
PL_LAN_Switch	PL	DHS	157.500	157.500
PL_Command_Measurement_Unit	PL	DHS	26.250	0.000
PL_Command_Control_Unit	PL	DHS	66.000	66.000
ESP_Command_Measurement_Unit	ESP	DHS	30.000	0.000
ESP_LAN_Switch	ESP	DHS	157.500	157.500
SE_LAN_Switch	SE	DHS	78.750	78.750
SE_Command_Measurement_Unit	SE	DHS	26.250	26.250
PL_LOS_Recorder	PL	DHS	33.000	0.000
SE_RCS_Thrusters	SE	Propulsion	1.737	0.120
SE_Latch_Valve_Biprop	SE	Propulsion	0.600	0.600
SE_Pyrovalves	SE	Propulsion	0.001	0.001
SE_RIT2X	SE	Propulsion	548.597	0.000
SE_RIT10_EVO	SE	Propulsion	83.223	0.000
SE_RIT2X_PCDU	SE	Propulsion	58.320	0.000
SE_RIT10_PCDU	SE	Propulsion	21.600	0.000
SE_RIT10_XRFS	SE	Propulsion	32.130	0.000
SE_RIT2X_XRFS	SE	Propulsion	32.130	0.000
SE_RIT10_FCU	SE	Propulsion	15.120	0.000
SE_RIT2X_FCU	SE	Propulsion	15.120	0.000
SE_Ion_Prop_Ancillary_HW	SE	Propulsion	12.000	0.000
SE_Main_Thrusters_400N	SE	Propulsion	2.738	2.100
PL_RCS_Thrusters	PL	Propulsion	2.432	0.336
ESP_RCU	ESP	Robotic	22.000	0.000
ESP_Manipulator	ESP	Robotic	16.500	0.000
ESP_Manipulator_Gripper	ESP	Robotic	2.200	0.000
PL_Manipulator_Control_Station	PL	Robotic	16.500	0.000
ESP_Manipulator_Rail_Mechanism	ESP	Robotic	3.000	0.000
SE_BetaRotator_Copy	SE	Structure	8.160	8.160
PL_DockingAdapter	PL	Structure	13.080	13.080
SE_AlphaJoint	SE	Structure	17.520	17.520
SE_BetaRotator	SE	Structure	8.160	8.160
PL_Airlock	PL	Structure	7.200	7.200
PL_Heat_Exchanger	PL	Thermal	18.480	18.480
PL_Heater_Controller	PL	Thermal	18.040	18.040
PL_Water_Pump_Module	PL	Thermal	103.200	103.200
SE_Heater_Controller	SE	Thermal	18.040	18.040
SE_Ammonia_Pump_Module	SE	Thermal	496.800	496.800
ESP_Heater_Controller	ESP	Thermal	18.040	18.040
PL_Dump_Device	PL	Venting	0.002	0.000
PL_Waste_Gas_Line_Shut_off_Valve	PL	Venting	0.000	0.002
PL_Repressurization_Valve	PL	Venting	0.002	0.000
PL_Venting_Pressure_Sensor	PL	Venting	1.575	0.000

12.1.1. Standard Mode Operation

For the standard mode operation, a maximum power consumption case and a minimum power consumption case are considered. As the maximum power consumption case, total power consumption excluding the power system is assumed to be 20 kW. As the minimum power consumption case, payload power consumption is assumed to be zero. For both cases, power consumption of the power system is estimated based on the power requirement from the other subsystems and it is added to the total power consumption.

Table 12-3: Maximum power consumption under the standard mode

	SE [W]	ESP [W]	PL [W]	Total [W]
Power consumption without Payload, and Power System	1882.561	425.040	1355.796	3663.397
Payload Power consumption	0	16336.600	0	16336.600
Power consumption without Power System	1882.561	16761.643	1355.796	20000.000
Power consumption of Power System	2911.125	4190.411	338.949	7440.485
Total power consumption	4793.686	20952.050	1694.745	27440.480

Table 12-4: Minimum power consumption under the standard mode

	SE [W]	ESP [W]	PL [W]	Total [W]
Power consumption without Payload, and Power System	1882.561	425.040	1355.796	3663.397
Payload Power consumption	0	0	0	0
Power consumption without Power System	1882.561	425.040	1355.796	3663.397
Power consumption of Power System	917.663	106.260	338.949	1362.872
Total power consumption	2800.224	531.300	1694.745	5026.269

The total power consumption is summarized in Table 12-3 and Table 12-4. Although the payload power consumption is allocated on the ESP, it can be placed in the PL as well. As the entire Free-Flyer system, maximum 27.44 kW and minimum 5.026 kW heat is necessary to be rejected to the space under the standard mode condition.

12.1.2. Survival Mode Operation

Under the survival mode operation, total power consumption will be lower than the minimum power consumption of the standard mode operation. As summarized in Table 12-5, the total power consumption excluding the power system is 1.556 kW and total 2.048 kW for the entire Free-Flyer system.

Table 12-5: Power Consumption under the Survival Mode

	SE [W]	ESP [W]	PL [W]	Total [W]
Power consumption without Power System	667.001	175.540	713.838	1556.379
Power consumption of Power System	269.144	43.885	178.460	491.488
Total power consumption	936.145	219.425	892.297	2047.867

12.2. Baseline Design

12.2.1. Heat Rejection Diagram

In order to handle large amount of heat generated by the Free-Flyer, an active thermal control system is required. The active thermal control system uses a mechanically pumped fluid in closed-loop circuits to perform heat collection, transportation and rejection. The overview of the active thermal control system is shown in Figure 12-1.

The pumped fluid system consists of two different loops: an external fluid loop and an internal fluid loop. The external fluid loop collects heat from each module: SE, ESP, and PL, and transfer heat to radiators and reject heat to the space. Working fluid of this system is ammonium, which is toxic but development and operational experience from the ISS system is available. On the other hand, the internal fluid loop is equipped inside PL module, collect heat from each component, transfer and reject heat to the external fluid loop via a heat exchanger. Working fluid of this system is water. Since the other two modules are not habitable areas, internal fluid loops are not required. For the SE module, cold plates are equipped to collect heat, and for the ESP module, fluid interfaces are equipped for payloads. Fluid control components, such as pumps, tanks are located in the SE module. Fluid pipes, heaters and thermal insulations are equipped all over the Free-Flyer, but detailed configuration is not defined in this study. There are two separate fluid loops to guarantee the redundancy of the active thermal control system.

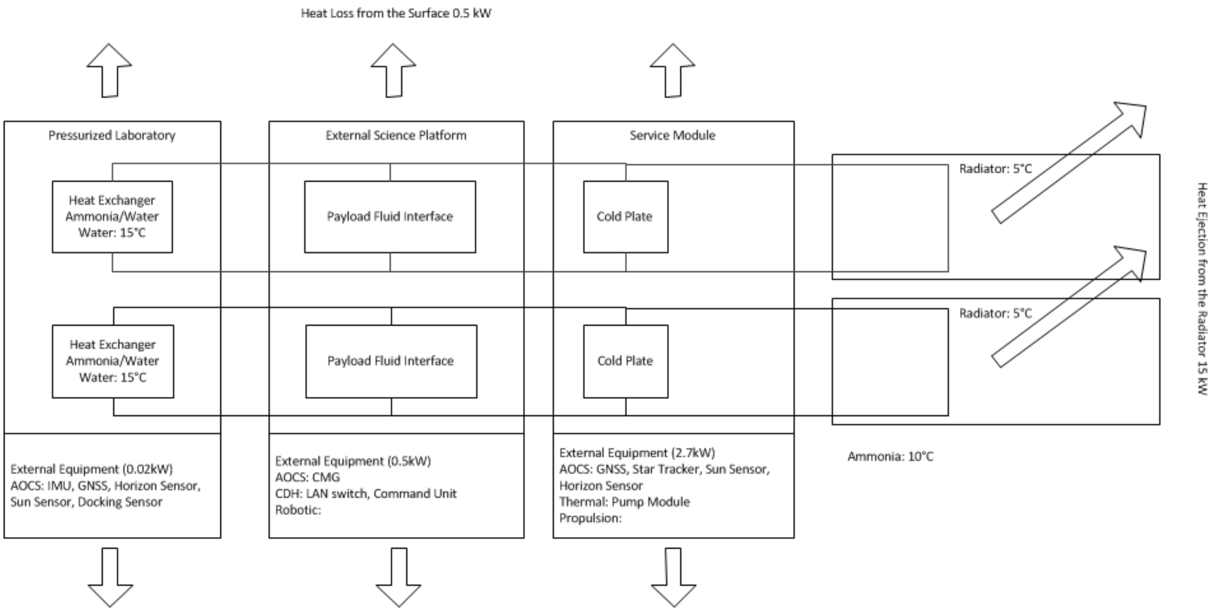


Figure 12-1: Layout of the baseline fluid loop of the thermal subsystem.

12.2.2. Heat Rejection Path

One main objective of this thermal study is sizing of the active thermal control system. As a whole system, the Free Flyer has several different heat rejection paths, such as:

- Surface Radiation through the Thermal Insulation
- Heat Rejection within the Payload System
- Heat rejection at body mounted radiators
- Heat rejection at the deployable radiator transported by the fluid loop

Therefore, the active thermal control system is not required to reject all the heat generated in Free-Flyer.

Surface Radiation through the Thermal Insulation

Part of generated heat is rejected through the thermal insulation. The amount of this surface radiation can be roughly estimated based on the following equation:

$$Surface\ Radiation = \varepsilon_{eff}\sigma A\left({T_{surface}}^4 - {T_{space}}^4\right)$$

$$A: surface\ area, 214m^2\ (PL: 80m^2, ESP: 67m^2, SE: 67m^2)$$
$$\varepsilon_{eff}: effective\ emissivity, 0.005$$
$$\sigma: stefan\ boltzman\ constant$$
$$T_{surface}: representative\ surface\ temperature$$
$$T_{space}: deep\ space\ temperature, 3K$$

Depending on the surface temperature and its distribution, the amount of surface radiation varies. Table 12-6 shows estimated surface radiation assuming uniform temperature over the surface. It should be noted this estimation does not consider the self-illumination within Free-Flyer.

Table 12-6: Estimation of the Surface Radiation

Tsurface [K]	Surface Radiation [W]
213	125
233	179
253	249
273	337
293	447

Body Mounted Radiator

Heat rejection occurs via body mounted radiators (including externally mounted component surfaces) as well. In order to save mass and size of the active thermal control system, the amount of heat rejection through the body mounted radiator is increased as much as possible, and the heat rejection requirement to the active thermal control system is minimized. Size of body mounted radiators is limited by the amount of heat rejection under the survival mode. While Free-Flyer's power consumption is low, the thermal system may not reject large amount of heat, otherwise a lot of survival heater power would be required. The active thermal control system has broad turn down capability for the heat rejection, by fluid control, radiator orientation control, and radiator retraction capability. On the other hand, body mounted radiators' turn down capability is limited. Therefore, the total amount of heat rejection through body mounted radiators shall not exceed power consumption under the survival mode condition. Considering the power consumption under the survival mode and estimated surface radiation through the thermal insulation, the maximum total area of body mounted radiators can be 12.4 m^2, and variation of heat rejection by radiator temperature is summarized in Table 12-7.

Table 12-7: Heat Rejection via Body Mounted Radiators

Tsurface [K]	Surface Radiation [W]
213	1302
233	1865
253	2593
273	3515
293	4664

Heat Rejection within the Payload System

Payloads can reject heat either to the fluid line or directly to the space. The amount of heat rejected directly to the space does not effect to the active thermal control system sizing. For this study, 15 kW heat rejections through the fluid loop, which includes payload and other subsystems, was assumed as a starting point. Under this assumption and the maximum power consumption condition, payloads need to reject 8542 W by itself, which is 52% of the total power, within the payloads' system. The following discussion basically relies on this assumption. However a design in which the payloads reject all the heat to the fluid line is mentioned as well.

Heat Rejection at the Deployable Radiator

The remaining heat shall be transported by the fluid loop and rejected to the space at the deployable radiator. Under the standard operation, the amount of heat rejection at the deployable radiator has to vary from almost zero to 15 kW. In order to reject 15 kW heat, the size of the deployable radiator shall be 25 m² with the average temperature of 5°C. On the other hand, in order to reduce the heat rejection when the total power consumption is small, the following measures are applicable

- Accepting the Sun input actively by controlling the radiator orientation. For this strategy, the absorbance of the radiator material should be higher than that of ISS radiator surface material Z-93 (ε=0.9, α=0.15) and the material selection requires further study considering weight, cost, optical property and stability under the space environment [RD 14].
- Allowing radiator freezing by freeze tolerant design of fluid pipes inside the radiator [RD 15]

12.3. Options and Trades

In the active thermal control system shown in Figure 12-1, fluid lines for each module are placed in parallel. Flow rate of each line can be controlled by valves, in order to fulfil the heat rejection requirement of each line. However, it would also be possible to change it to a different configuration in which different modules or some of the components in different modules are connected in series. Since all three modules of Free Flyer are planned to be launched in an integrated configuration, it is reasonable to optimize the fluid line configuration for the entire Free Flyer. Possible influences are listed below:

- By placing heat loads in an efficient order, fluid temperature at the end of the fluid loop can be increased, which reduces required radiator size.
- By connecting different modules, fluid loop design and flow control become more complicated

For the quantitative evaluation, it is necessary to model the possible active thermal control system configuration.

12.4. Mass and Power Budget

Mass, size and power of the thermal system components is estimated based on currently used ISS component information and resized for the heat rejection requirement of Free Flyer [RD 15] [RD 16] [RD 17]. Therefore, each component technology is available, but the real components are necessary to be designed in the future.

12.4.1. List of Equipment

Each component mass is estimated based on the assumption that the amount heat rejection on the active thermal control system is 15 kW, which requires payloads to reject part of their heat within the payload system. When the heat rejection requirement is increased to 23.542 kW, the estimated mass will be increased as well. For the 15 kW heat rejection case, total mass of thermal system is estimated to be 1666 kg without margin and 1873 kg with margin. In case of 23.542 kW heat rejection requirements, the estimation is to be increased to 2169 kg without margin and 2448 kg with margin.

Table 12-8: Mass budget of the thermal equipment (15kW rejection case)

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Ammonia_Coolant_Pipe	160.00	20.00	32.00	192.00	10.25
ESP_External_Cold_Plate	0.00	CalcErr	0.00	0.00	0.00
ESP_Heater	10.00	10.00	1.00	11.00	0.59
ESP_Heater_Controller	60.00	10.00	6.00	66.00	3.52
ESP_MLI	72.00	10.00	7.20	79.20	4.23
PL_Heat_Exchanger	80.00	10.00	8.00	88.00	4.70
PL_Heater	10.00	10.00	1.00	11.00	0.59
PL_Heater_Controller	60.00	10.00	6.00	66.00	3.52
PL_Internal_Cold_Plate	80.00	10.00	8.00	88.00	4.70
PL_MLI	80.00	10.00	8.00	88.00	4.70
PL_Water_Coolant_Pipe	20.00	20.00	4.00	24.00	1.28
PL_Water_Pump_Module	120.00	20.00	24.00	144.00	7.69
SE_Ammonia_Pump_Module	100.00	20.00	20.00	120.00	6.41
SE_Ammonia_Tank_Assembly	120.00	10.00	12.00	132.00	7.05
SE_External_Cold_Plate	80.00	10.00	8.00	88.00	4.70
SE_Heater	10.00	10.00	1.00	11.00	0.59
SE_Heater_Controller	60.00	10.00	6.00	66.00	3.52
SE_MLI	94.00	10.00	9.40	103.40	5.52
SE_Nitrogen_Tank_Assembly	90.00	10.00	9.00	99.00	5.29
SE_Radiator	360.00	10.00	36.00	396.00	21.15

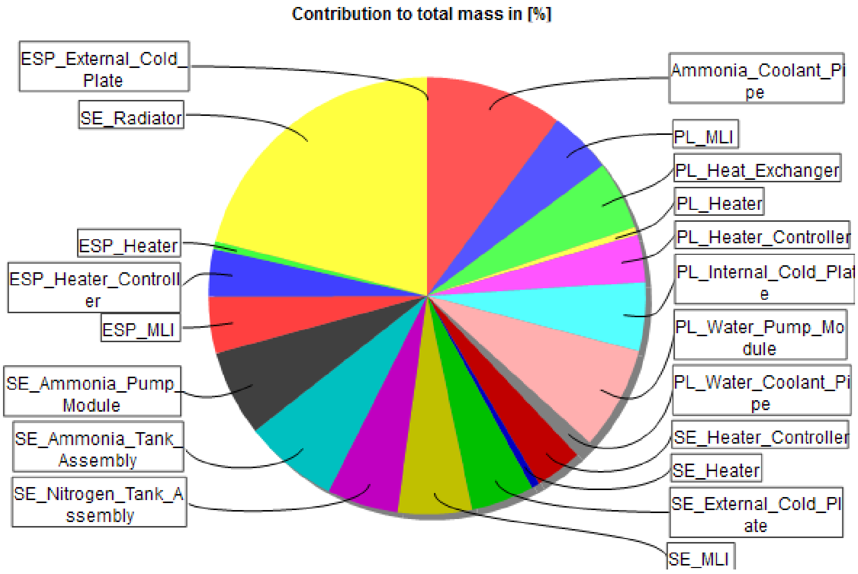


Figure 12-2: Pie chart of mass distribution of the thermal equipment

12.4.2. Power Budget

Same as the mass estimation, Table 12-9 shows power budget for the design of 15 kW heat rejection case. For this case, total power of thermal system is estimated to be 673 W for the standard mode. In case of 23.542 kW heat rejection requirements, the estimation is to be increased to 993 W for the standard mode.

Table 12-9: Power budget of the thermal subsystem.

	Power avg standard [W]
PL_Heat_Exchanger	18.480
PL_Heater_Controller	18.040
PL_Water_Pump_Module	103.200
SE_Heater_Controller	18.040
SE_Ammonia_Pump_Module	496.800
ESP_Heater_Controller	18.040

12.4.3. Mode dependencies

Under the survival mode condition, reduced Free-Flyer's power consumption causes low temperature level of the entire Free Flyer. The thermal system has to reduce the heat rejection, and protect each component from over cooling by using survival heaters. To reduce the heat rejection of the active thermal control system, the following measures are possible:

- Flow rate control of the fluid loop
- Freezing tolerant design in the radiator fluid pipe
- Radiator orientation control
- Radiator retraction

Depending on the cause of the survival mode, for example attitude control failure or alpha joint control failure, radiator orientation control could be difficult. Therefore, a suggestive measure under the survival mode is to reduce the flow rate and reducing heat rejection from the radiator by radiator retraction or allowing the part of radiator freezing. Low heat rejection requirement under the survival mode can be covered by heat rejection from body mounted radiators and minimized heat rejection from the active thermal control system. Survival heater operation is necessary to avoid any local over cooling, especially for components which are mounted near the Free-Flyer external surface.

12.5. Re-Supply Items

For the thermal system operation, no re-supply item is planned.

12.6. To Be Further Studied / Additional Considerations

In this study, overall active thermal control system configuration has been proposed, and first mass and power estimation were presented. For the next study phase, the system feasibility shall be studied quantitatively including the following points:

- Specifying environmental input considering Free-Flyer orbit, Free-Flyer attitude control and Free-Flyer external shape
- Modelling of fluid loop considering location and power consumption profile of components of Free-Flyer
- Feasibility study of the optimized fluid loop design

12.7. Summary / Comparison to ISS

Table 12-10 shows a summary of sizing results of Free-Flyer and information of ISS External Active Thermal Control System (EATCS) for comparison [RD 15]. Although nominal heat rejection of the ISS EATCS is 70 kW, the size of its radiator is much larger than that of necessary size for this nominal heat rejection. This design would allow ISS system to perform higher power operation than nominal and extension flexibility in the future. For the Free-Flyer case, a heat rejection requirement is to be specified into narrower range already during the design phase, which makes Free-Flyer system compact and adjusted for expected mission.

Table 12-10: Summary of TCS Design

	Free-Flyer TCS (15kW rejection case)	**Free-Flyer TCS (23.5kW rejection case)**	**ISS EATCS**
Heat Rejection via Active TCS	Max 15 kW	Max 23.542 kW	70 kW or more
Radiator size	25 m²	38 m²	475 m²
Radiator weight	354 kg	538 kg	6732 kg

13. Structure

13.1. Requirements and Design Drivers

Free-Flyer comprises 3 parts:

- Pressurised Lab
    - 12 ISPRs
    - Man-Tended
    - IBDM
- External Science Platform
    - 10 JEM EF PL
    - 8 m² for smaller payload
    - Transfer launch loads to ESP and PL
- Service Module
    - Accommodates propulsion, power, comms
    - Launch adapter
    - Transfer launch loads to ESP

13.2. Baseline Design

Transfer loads from launcher to vehicle

- avoid changes in shape/ diameter as much as possible
    - ESP and SE identical shape
    - ESP and PL similar in diameter
    - Large launch adapter (probably dedicated development required)

Present Assumption

- Classical design = metal

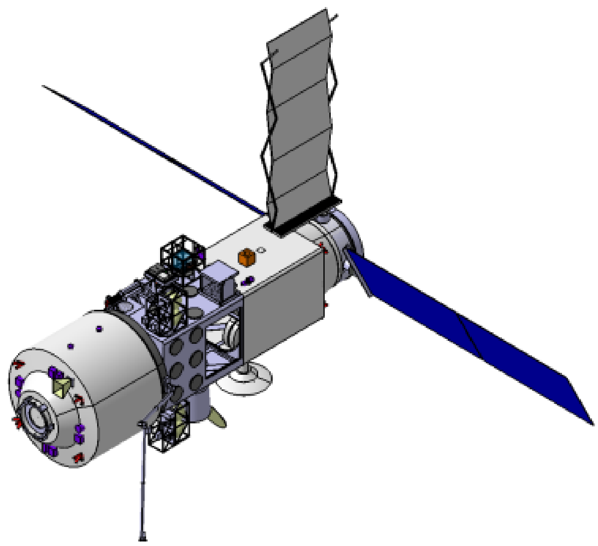


Figure 13-1: Shape of Free-Flyer

Total Mass: 5110 kg (6100 kg)
(Remark: mass savings may be possible by making use of CRFP)

PL Comprises:

- IBDM
- Pressurised Volume (takes launch loads)
    - 4 ISPRs for Launch
    - 8 super lightweight racks
- Airlock
    - Data from JEM used
    - Partly new development to account for change in accommodation
- MMOD (micro-meteoroid orbital debris shield)

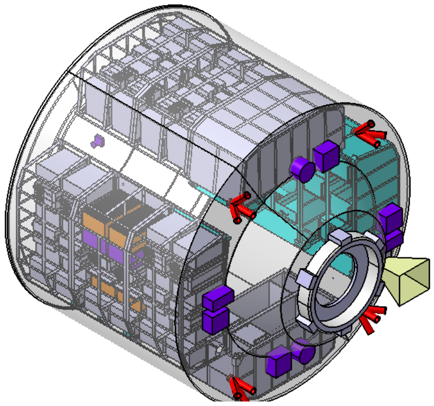


Figure 13-2: Pressurised Lab concept

Total Mass: 3000 kg

ESP Comprises:

- Box-Structure to transfer loads and accommodates payload on 3.5 sides
    - Struts and shear walls
- Interfaces to payload
- Airlock and CMGs
- Adapter to Pressurized lab accommodating robot race ring

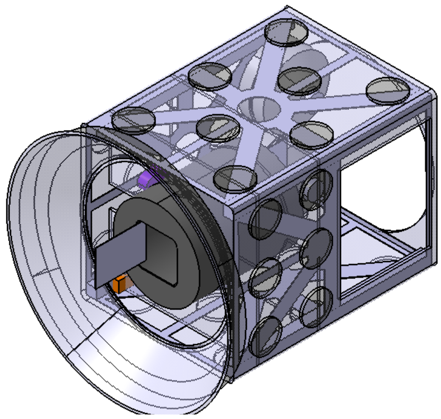


Figure 13-3: Structure concept of External Payload Platform

Total Mass: 740 Kg

Service Module Comprises:

- Box-Structure to transfer loads and accommodate subsystems
    - Struts and shear walls
- Circular ring at the end of SE to accommodate
    - beta-ring and interface to radiator, solar arrays
- Adapter to Launch Adapter Ring

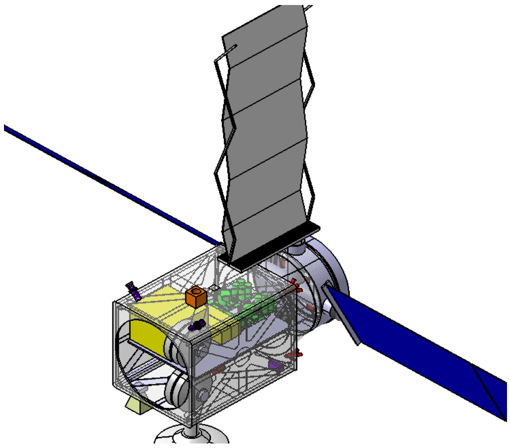


Figure 13-4: Structure concept of Service Module

Total Weight: 1270 kg

13.2.1. Radiation Shielding

LEO environment:

- High-energy protons and electrons,
- High energy and charge nuclei,
- X- and Gamma- radiation,
- μ-meteorites, space debris,
- ATOX,
- UV and VUV.

Radiation Shield:
Radiation shield is a structure which absorbs corpuscular- and electromagnetic radiation → effectively decreasing energy deposited within astronauts' tissues and electronic parts

- Spacecraft's hull,
- MLI, Debris shilding,
- Support hardware,
- Payloads,
- Storage of water,
- Food,
- Clothes,
- …

ALTEA detector (measurements from 2006 - present) ≲ 1.5 nSv/s [1] = **0.13 mSv/d**

Space Radiation Analysis Group, Johnson Space Center

Organ specific exposure limits for astronauts

Exposure Interval	Blood forming organs	Eye	Skin
30 days	25 rem	100 rem	150 rem

25 rem/30 days = 0.25 Sv/30 days = 8.(3) mSv/d

The hydrogen based materials, e.g. Polyethylene and Kevlar are the most effective materials of stopping the so-called secondary particle production.

Figure 13-5: Radiation shielding materials (left: Polyethylene ρ=0.9 g cm^{-3} ; right: Kevlar ρ=1.4 g cm^{-3})

- A thickness of approx. 5 cm Polyethylene (4.8 g cm-2) material will cause a mass input of ~2 tons to the Pressurized Laboratory segment

- The drop of the equivalent radiation dose is then
  - ~10% [3] (for 4.8 g cm-2 – model calculation for Zvezda)
  - ~20% [2] (for 5.0 g cm-2 – measured value on Columbus)

- Due to the fact that crew member(s) will work occasionally in the Pressurized Laboratory module the additional radiation shielding is not required.

[RD 1][RD 2][RD 3]

13.2.2. Debris Shielding

Requirements and Assumptions:

The majority of the space debris objects hit a spacecraft frontal in flight direction as well as on port/starboard sides. This statement is supported by simulation results using existing environmental models e.g. MASTER (ESA) and ORDEM (NASA) as well as by ISS experience. The critical components of the current ISS are shielded against objects with a diameter of 1 cm at a velocity of 9 km/s.

> "*Most critical components in the velocity ram and port/starboard sides are shielded to **1 cm** debris at typical impact velocities of **9 km/s** and impact angles of **45 deg**.*" *[RD 18]*

Figure 13-6 shows the most affected areas of the ISS (red coloured). An object of 1 cm in diameter at a velocity of 9 km/s has a kinetic energy of 57 kJ. The energy can be subdivided into orthogonal and parallel portion (45° impact angle), which results in a kinetic energy of ca. 30 kJ. The shielding concept of the European Columbus module was verified by Hypervelocity Impact (HVI) at Fraunhofer Institute for High-Speed Dynamics, Ernst-Mach-Institute (EMI) in Freiburg, Germany. The shielding concept was tested by projectile with a diameter of 15 mm and a relative velocity of 7 km/s.

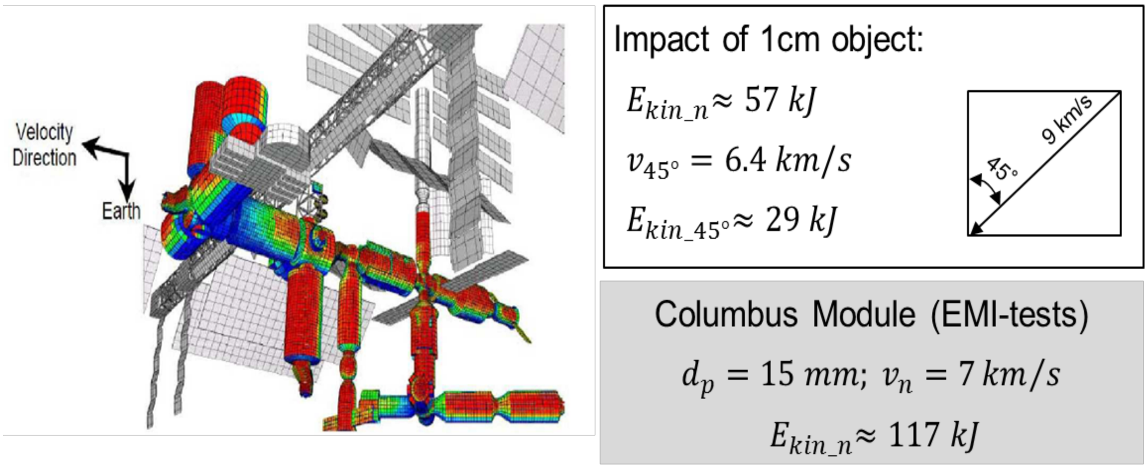


Figure 13-6: Left: Impact flux into ISS (red: high number of impacts, blue: low number of impacts) [RD 18], right: kinetic energy estimation of 1 cm object (top), HVI-test parameter for Columbus module (bottom).

Using the kinetic energy threshold value of Columbus shielding concept a range of object diameters at different velocities can be defined. Figure 13-7 shows different object diameters at different velocities. At the ISS orbit, the space debris impacts occur mainly at a velocity of 10 km/s. Micrometeoroids predominantly have a relative velocity of 20 km/s. This means that objects with a diameter larger than 6 mm need to be considered for the shielding design.

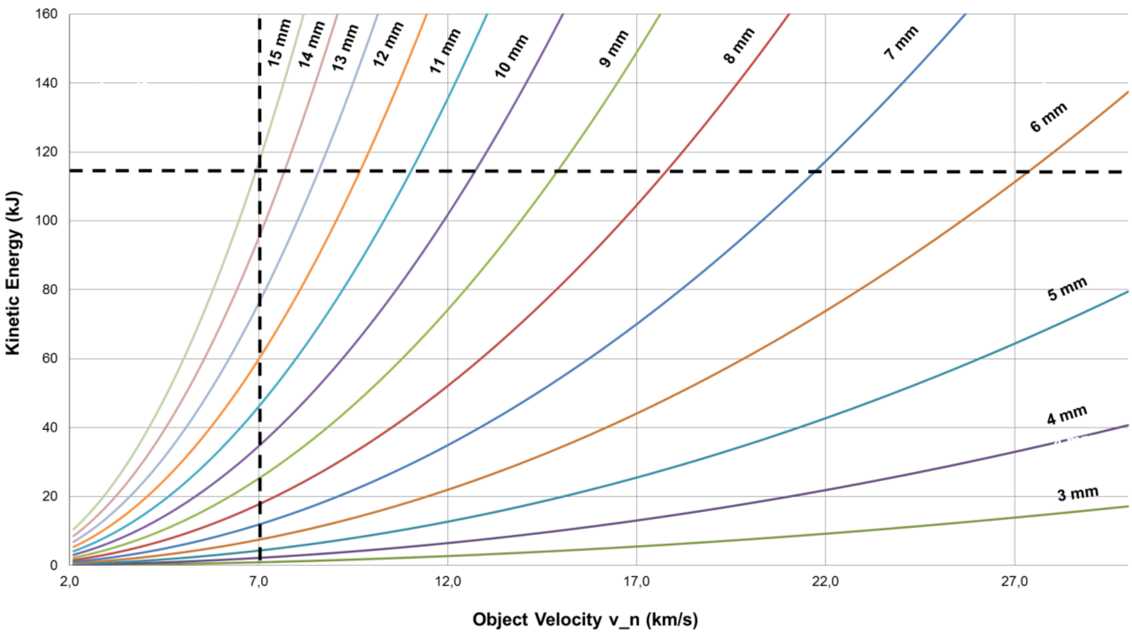


Figure 13-7: Kinetic energy of impact particles

Different types of shielding concepts were developed and utilized to the ISS modules in the past. Figure 13-8 (left top) shows a double wall system, the so called Whipple shield, as an example. The shielding consists of a thin bumper separated by a distance (standoff) to a rear wall. The main idea here is to distribute the kinetic energy of the impacting object (debris cloud) over a large area and to reduce the loads to the rear wall (see Figure 13-8 left bottom). This can be done by variation of thickness of the bumper, rear wall or more effectively by a large standoff. Figure 13-8 (right) shows a ballistic limit curves for a fixed standoff (S) and rear wall thickness (tw) but different bumper thicknesses (ts). An increasing thickness of the bumper wall offers a higher safety against impacting objects.

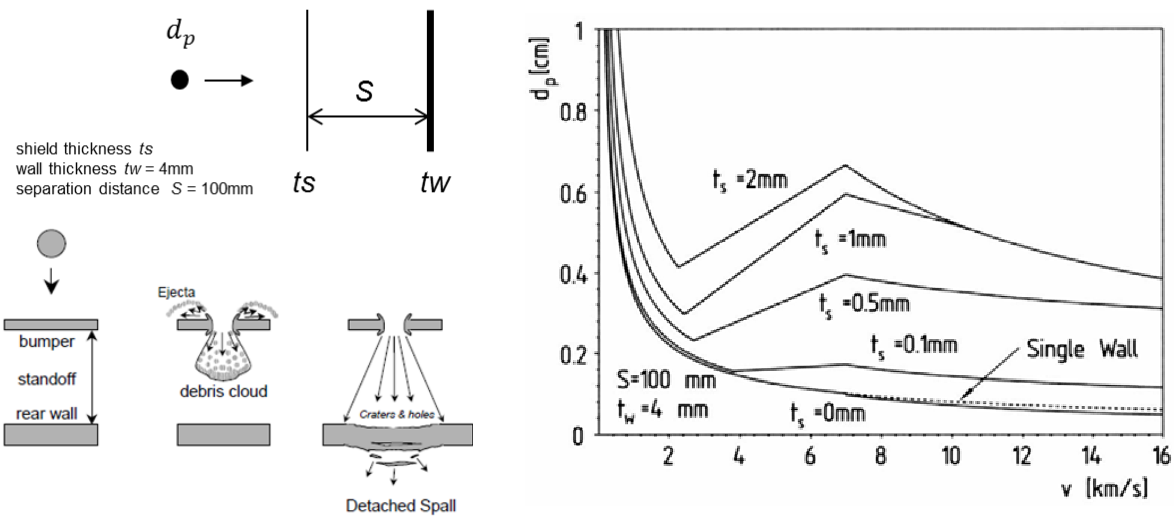


Figure 13-8: Left top: Whipple shield; left bottom: schematic principle of impact process; right: ballistic limit curves for Whipple shielding

Figure 13-9 shows different shielding concepts of NASA, NASDA (today JAXA) and ESA (top) and the corresponding ballistic limit curves (bottom). The shielding systems protect the ISS against impacting objects at a relative velocity of ca. 7 km/s most effectively.

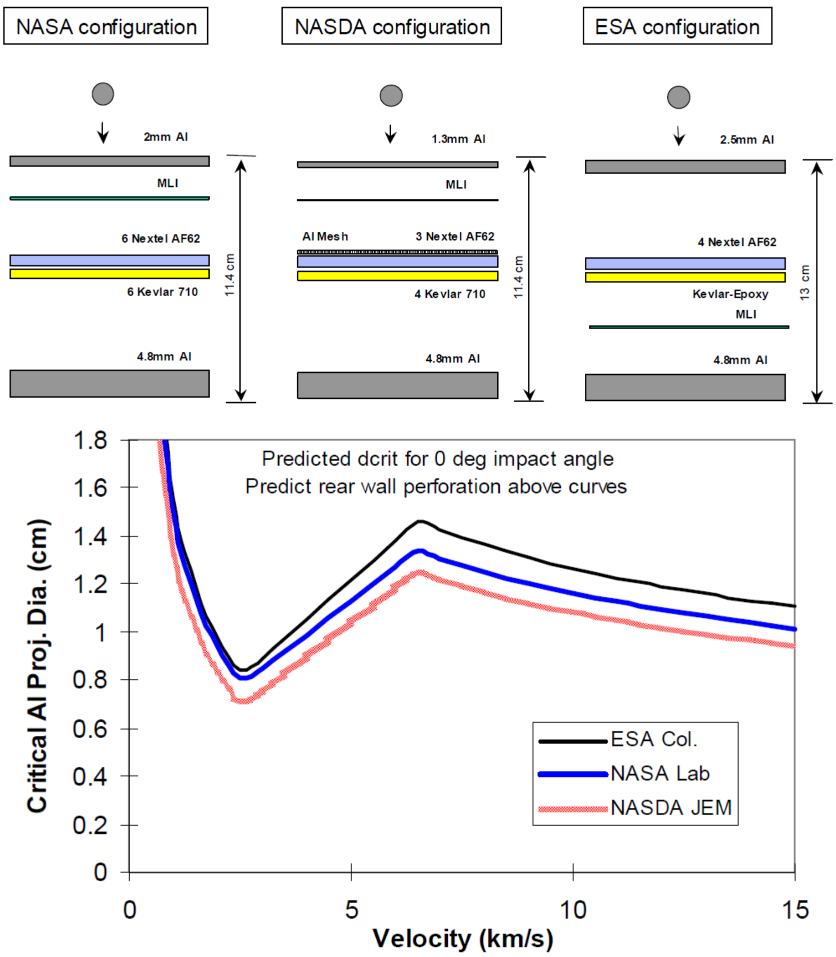


Figure 13-9: top: Nextel/Kevlar enhanced Whipple shield configuration on ISS; bottom ISS Nextel/Kevlar stuffed Whipple shield ballistic limits [RD 19]

As outlined, the most critical hardware of the ISS is shielded against objects with a diameter in the range of d_p=1 cm to 1.3 cm at an impact velocity of v_p=9 km/s and impact angle of 45°. Under similar impact conditions the former space systems were able to stop much smaller objects. Hereafter some examples of already flown shielding systems are listed [RD 19]:

- Mir space station: 0.3 cm [RD 19]
- Space Shuttle Orbiter: 0.2 cm to 0.5 cm [RD 19]
- Apollo and Skylab: 0.15 cm to 0.2 cm [RD 19]

85% of the ISS is shielded with an average mass/area ratio of 10 kg/m². The „average" shielding capability of the ISS modules is 1cm [RD 20].

Experts worldwide are working on new concepts to improve the efficiency of the shielding technologies. In the frame of the ReVus project [RD 21] the Fraunhofer EMI performed HVI-test on the so called "B7-B" shielding concept. This shielding has a mass/area ration of ca. 7.4 kg/m² (ca. 26% lower than current ISS shielding) and is able to protect modules against projectiles with a diameter of 4.5 mm at a relative velocity of 7 km/s [RD 21]. Figure 13-10 shows the shielding concept (left top), the HVI-tested prototype (top right) and the ballistic limit curve for this shielding system (bottom).

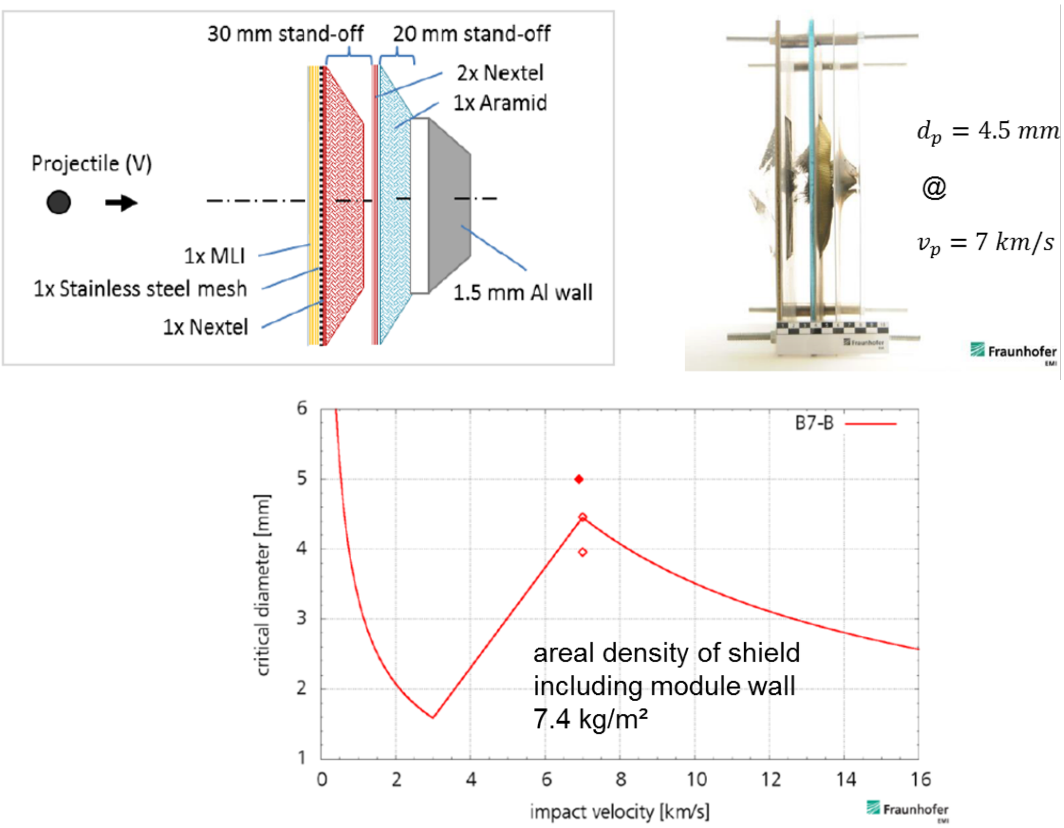


Figure 13-10: new shielding concept (B7-B) tested by Fraunhofer EMI in the frame of ReVus project [RD 21]

The Post-ISS Free-Flyer debris and micrometeoroids analysis is performed by utilizing the ESA-MASTER2009 model. The settings for this analysis are listed below:

- Orbit: h=400 km; i=51.6; e= 0.0006
- Operational time: 15 years
- Considered objects: diameter 5 mm – 10 m
- Future scenario: business as usual
- Debris sources: all
- Meteoroid sources: all
- Meteoroid Streams: Jenniskens-McBride
- Target type: sphere

Figure 13-11 shows the impact velocity distribution of space debris and micrometeoroids at the ISS orbit. The space debris impacts occur predominantly with 10 km/s and micrometeoroids with 20 km/s.

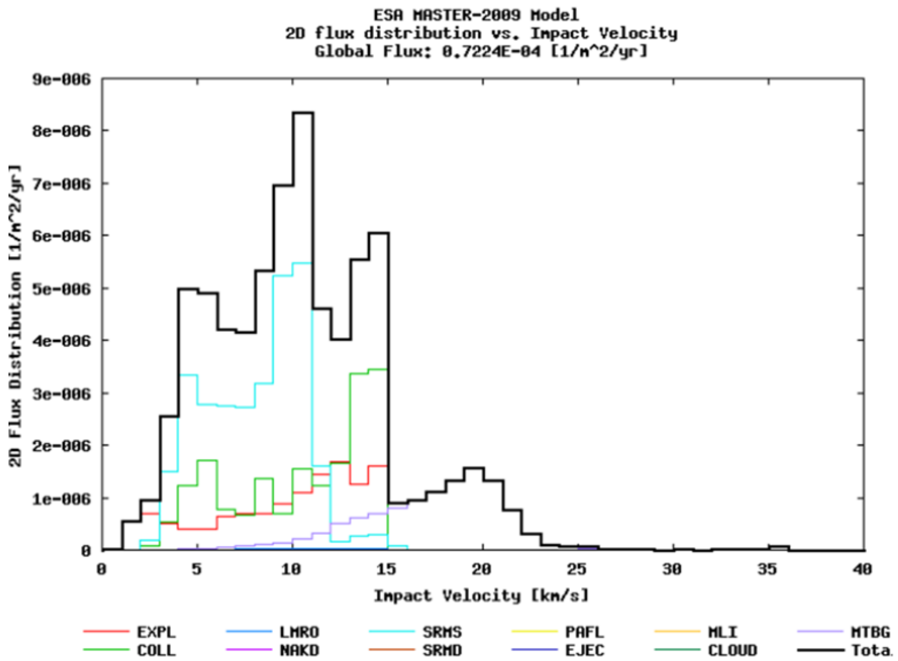


Figure 13-11: Impact velocity distribution of space debris and micrometeoroids, analysis tool: MASTER 2009

Figure 13-12 illustrates the expected impact elevation. Most impacts take place at an elevation of zero degree.

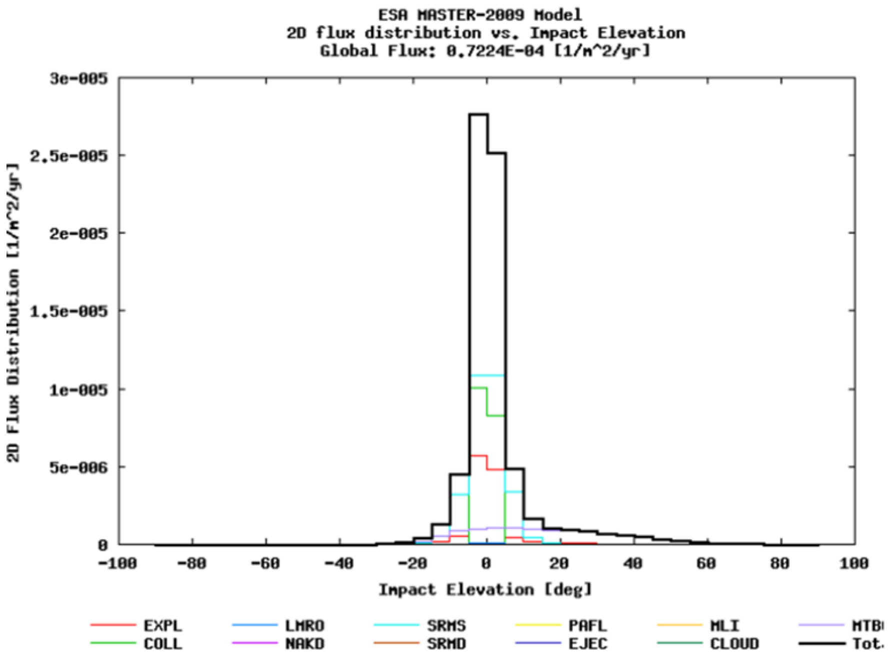


Figure 13-12: Impact elevation distribution of space debris and micrometeoroids, analysis tool: MASTER 2009

The impact azimuth is shown in Figure 13-13. The impacts occur predominantly by indicated angles.

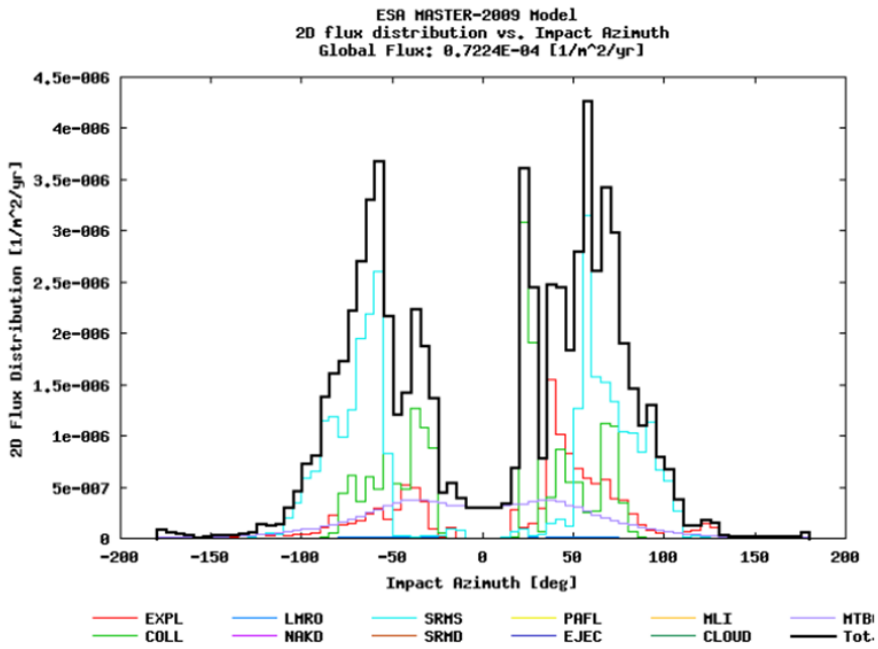


Figure 13-13: impact azimuth distribution of space debris and micrometeoroids, analysis tool: MASTER 2009

Figure 13-14 shows the baseline design of the Post-ISS Free-Flyer that has been utilized for the shielding analysis. The pressurized laboratory (PL) has a total area of ca. 80 m² and the service module (SE) of ca. 67 m². Considering the simplified analysis performed with MASTER2009 (target type: sphere), the total area of the Free-Flyer (ca. 150 m²) is divided by factor of 4. This results in a total area of ca. 38 m² (PL + SE) and was used for the analysis.

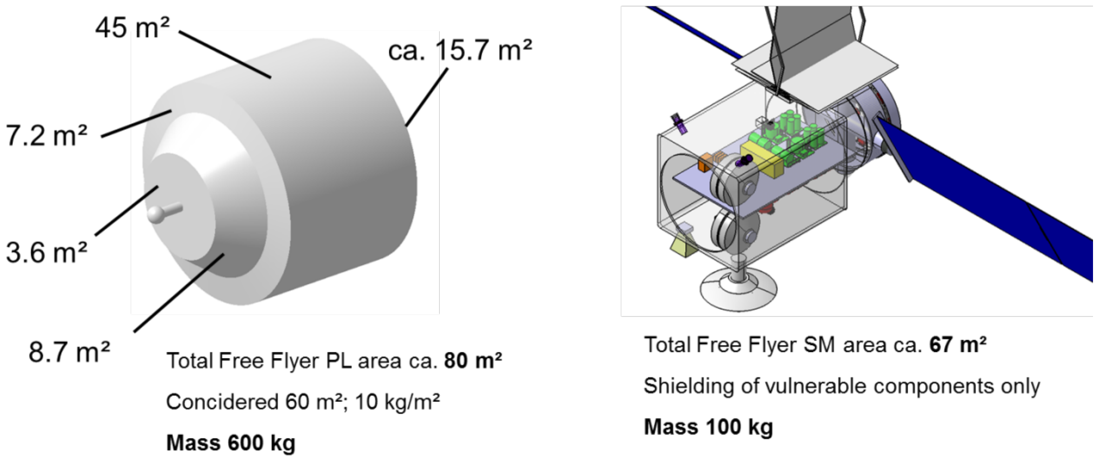


Figure 13-14: Baseline design of the Free Flyer's debris protection.

Since only a minor number of impacts can be expected on the back side of the PL (see Figure 13-14 ca. 16 m²) an area of 60 m² was taken into account for the mass estimation of shielding system. An area/mass ratio of 10 kg/m² was chosen, as it currently is realized on ISS. The SE is assumed to be less vulnerable than the PL. This assumption is supported by the fact that even if an impacting particle penetrates the structure wall of the SE, this will not necessarily leads to ta system failure and corresponding loss of the mission. Therefore for the SE a local shielding only is considered for protection of vulnerable components. The SE shielding mass is estimated to be ca. 100 kg.

Table 13-1 summarizes the results of the space debris and micrometeoroids impact analysis. The table shows the total flux as well as the expected total number of impacts. Furthermore the probability of no penetration (PNP) and the risk of penetration (R) are calculated.

Table 13-1: Summary of impact flux, expected impact rates, PNP and risk

Diameter	Total flux	MM flux	Expected impacts	PL & SM area	Mission duration	PNP	R
dp	**F**		**N**	**A**	**delta_t**		
(mm)	(1/m²/Jahr)	(%)		(m²)	(years)		(%)
5	5,83E-05	14%	3,32E-02	38	15	0,96733	3,3%
6	3,86E-05	8%	2,20E-02	38	15	0,97822	2,2%
7	3,67E-05	4%	2,09E-02	38	15	0,97932	2,1%
8	1,48E-05	5%	8,44E-03	38	15	0,99159	0,8%
9	9,88E-06	4%	5,63E-03	38	15	0,99438	0,6%
10	2,09E-06	13%	1,19E-03	38	15	0,99881	0,1%

The historical meteoroid/debris shielding requirement for the ISS is 0.98 to 0.998 per critical element over 10 years [RD 19]. The estimated PNP for 15 years (also true for 10 years) is lower than the ISS numbers. The analysis was performed by applying a number of simplifications such as: consideration of all objects with a diameter 5 mm – 10 m, Free-Flyer (target) is considered as a spherical object with a cross section area of 38 m², all impacting objects have sufficient energy to penetrate the wall, no distinction between vulnerable and not vulnerable components (all components are assumed to be vulnerable in case of impact). Therefore, the following issues need to be studied in more detail in the frame of the project:

- Geometry of modules and shadowing aspects (e.g. shadowing effects of solar panels, radiator, main station by using e.g. ESABASE)
- Accommodation of components and corresponding vulnerabilities (using e.g. PIRAT)
- Research regarding the availability of new and more effective shielding systems
- Optimization of shielding on PL and SE (application of different shielding)
- Hazard to payload and robotic manipulator

Note: the MASTER-model simulation results represent a statistical probability only, a critical event can occur any time of the mission. Therefore a shielding system for the Free-Flyer is recommended.

13.3. Mass Budget

13.3.1. List of Equipment

Table 13-2: Mass budget of the structure domain.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
ESP_PayloadInterface	240.00	20.00	48.00	288.00	4.72
ESP_PrimaryStructure	500.00	20.00	100.00	600.00	9.84
PL_Airlock	300.00	20.00	60.00	360.00	5.90
PL_DockingAdapter	340.00	20.00	68.00	408.00	6.69
PL_MMOD	600.00	20.00	120.00	720.00	11.80
PL_ModulStructure	1000.00	20.00	200.00	1200.00	19.67
PL_RackForLaunch	320.00	10.00	32.00	352.00	5.77
PL_RackForOrbit	240.00	20.00	48.00	288.00	4.72
PL_SecondaryStructure	200.00	20.00	40.00	240.00	3.93
SE_AlphaJoint	120.00	20.00	24.00	144.00	2.36
SE_BetaRotator	250.00	20.00	50.00	300.00	4.92
SE_LaunchAdaptRingAttach	200.00	20.00	40.00	240.00	3.93
SE_MMOD	100.00	20.00	20.00	120.00	1.97
SE_PrimaryStructure	500.00	20.00	100.00	600.00	9.84
SE_SecondaryStructure	200.00	20.00	40.00	240.00	3.93

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	5110.00			6100.00	

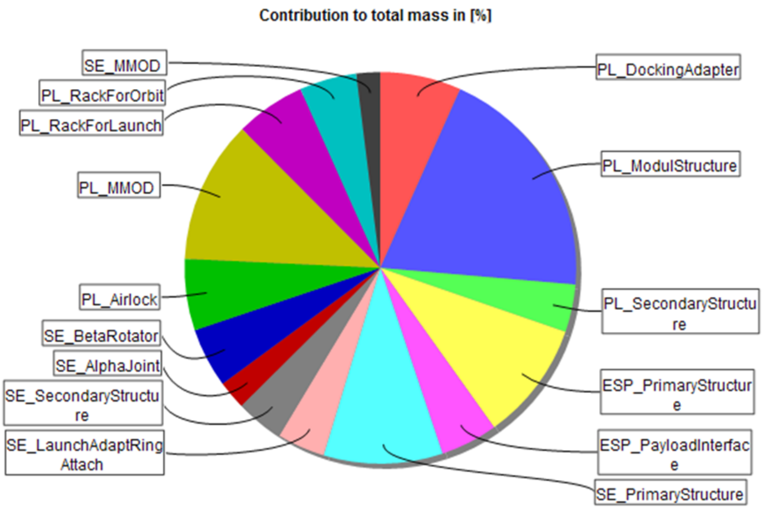


Figure 13-15: Pie chart of mass distribution of the structure parts.

13.4. To Be Further Studied / Additional Considerations

- Detailed design of structure
- Launcher adapter
- Potential mass savings using CFRP

14. Robotics

14.1. Requirements and Design Drivers

The main requirements for the robotic manipulator are the following:

- Arm shall be used to pick experiments from payload (PL) airlock and place them on an arbitrary spot on the experiment platform
- No arm for capsule/vehicle berthing – docking shall be autonomous

Advancing from the baseline design as shown in Figure 14-1, during the course of the study, the experiment platform became cubed in shape for structural reasons with the airlock being located in the middle of this structure, cp. Figure 14-2. Hence, a new approach had to be found as a replacement for the linear rail system as planned within the baseline design.

14.2. Baseline Design

The baseline design comprised a 10 m-long arm mounted on a linear rail for reaching all areas of the platform and the air lock at Bigelow module. The major problem here was the poor reachability of lower platform side.

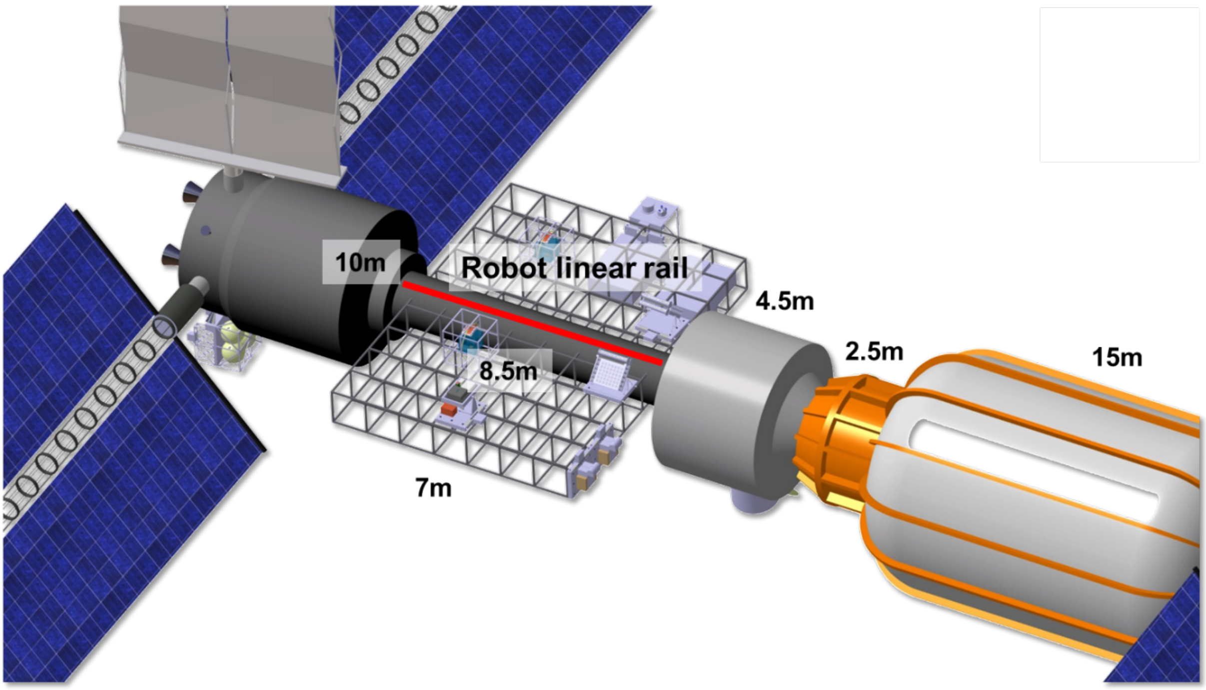


Figure 14-1: Historic robotic concept from Base Station CE-study.

By moving the airlock to the pressurized lab instead of the Bigelow, a cube-sized experiment platform design became favourable, mainly for structural reasons. By mounting the arm on a circular rail system located at the edge of the pressurized lab, full reachability of the experiment platform could be achieved while giving access to the airlock.

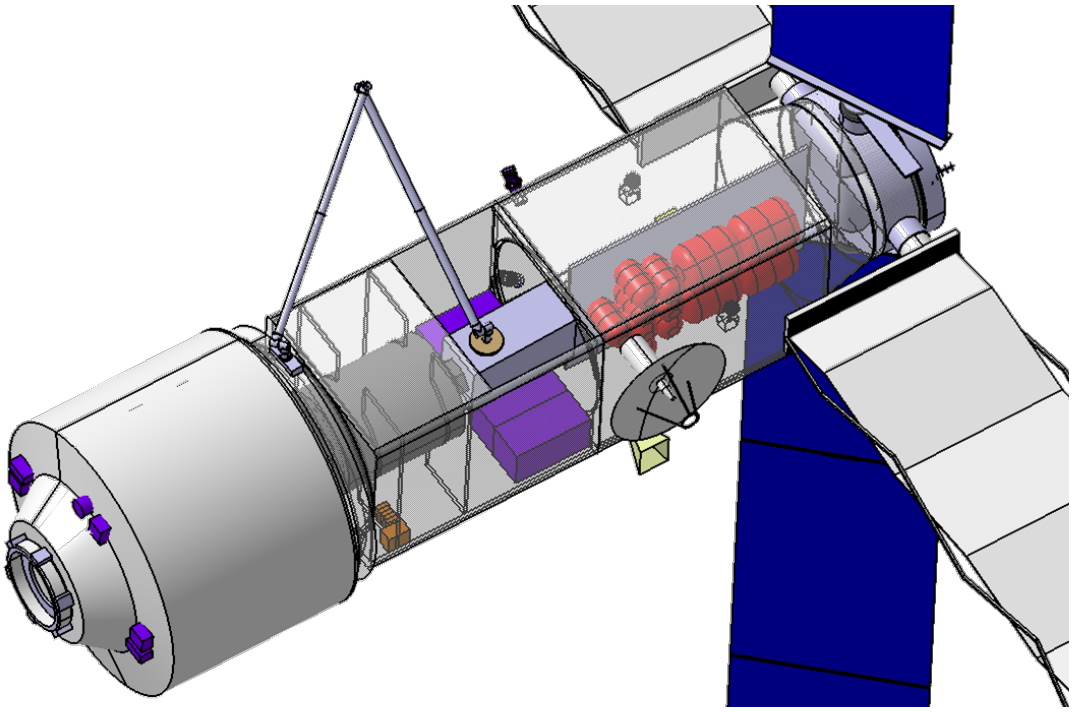


Figure 14-2: Baseline design of the robotic arm.

The proposed configuration of the robotic manipulator features a total length of 9.7 m, see Figure 14-3. In its stowed configuration, the arm consumes approximately half that length. It has seven joints set up in an alternating roll-pitch configuration. A stereo-camera system at the arm wrist is used for visual servoing in order to achieve a greater pointing accuracy of the tool center point (TCP). The maximal nominal torque accounts to 160 Nm. The arm features an integrated joint design with position, and torque sensors as well as a break. Thus, while giving security in case of power outages, a high-precision position and force-sensitive impedance control can be achieved for force-sensitive and environment-compliant grasp and manipulation tasks. In addition, two fully redundant mechatronic strings are integrated that can be switched in a cold redundancy scheme. The arm can be controlled in an autonomous mode; it further supports shared autonomy and a telepresence mode.

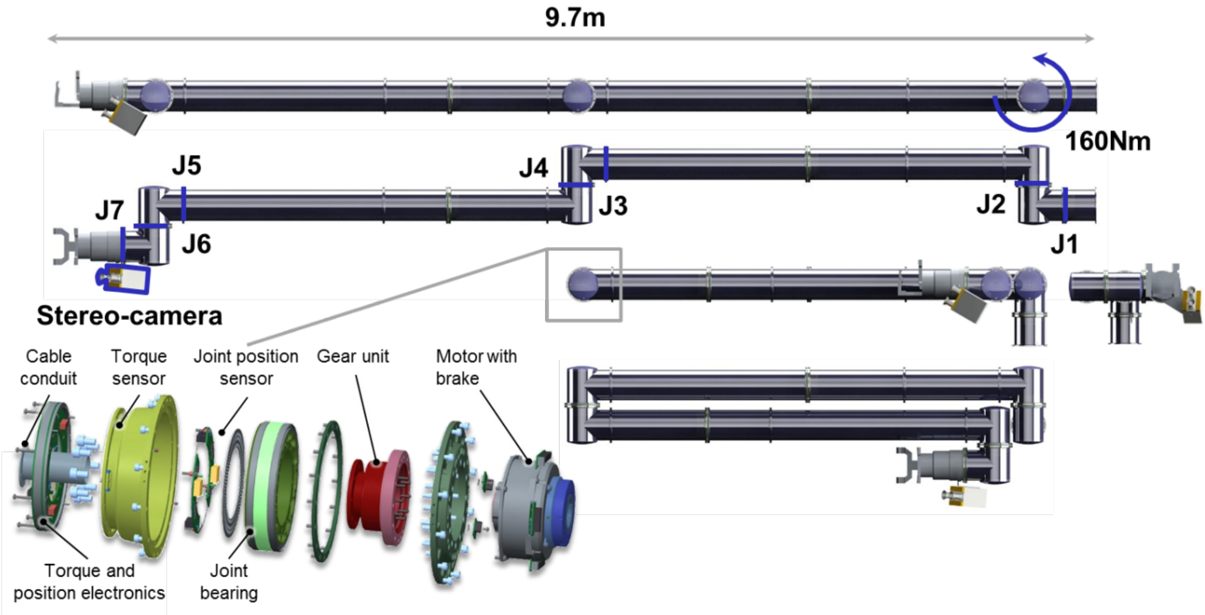


Figure 14-3: Robotic arm configuration.

Figure 14-4 depicts the Free-Flyer with a visiting vehicle docked to the docking node. During the study the team opted for an autonomous docking manoeuvre of the approaching vehicle not including the robotic manipulator in docking operations. Given the current length of the arm this would also not be supported at the moment. For supporting berthing using the robotic arm, a much longer design in the order of at least 15 m as given for the Canadarm aboard ISS would be required.

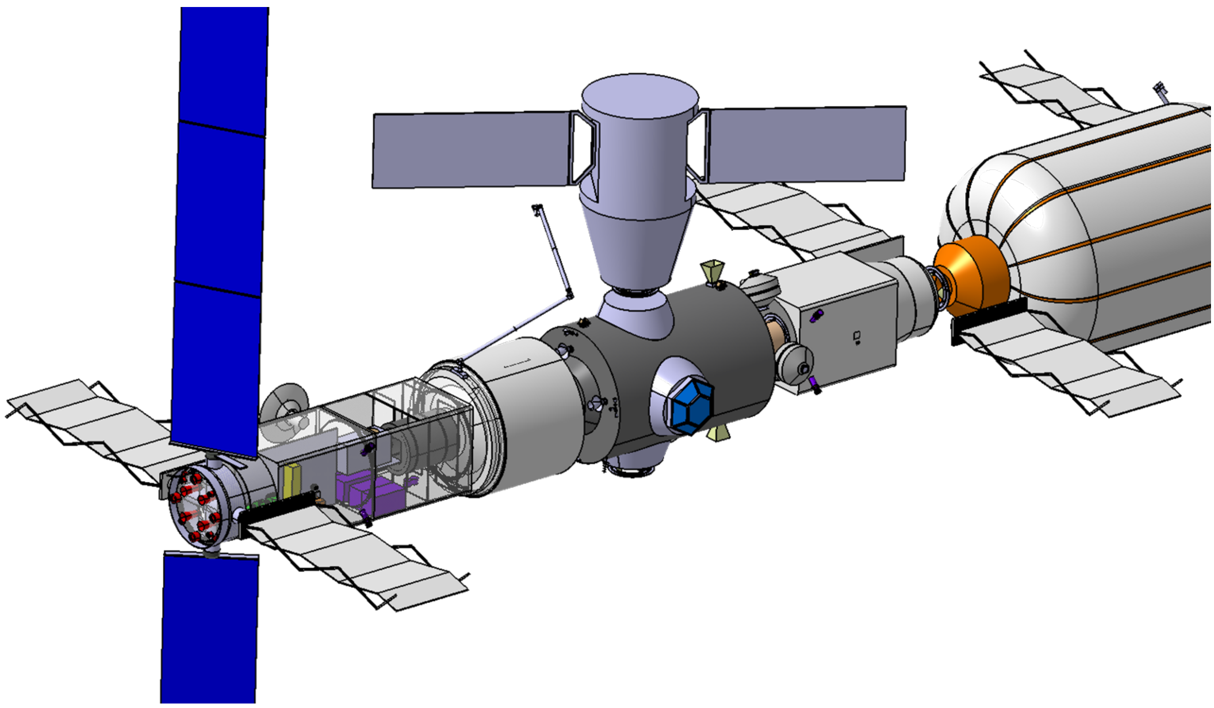


Figure 14-4: Robotic arm reachability restrictions.

Table 14-1: Robotic arm geometries.

total arm length [m]			**9,706**	
long segments difference [m]			**0,140**	
a [m]	**alpha [deg]**	**theta [deg]**	**d [m]**	**type**
0	0	0	1,756	roll
0	-90	0	0,168	pitch
0	90	180	3,870	roll
0	-90	0	0,168	pitch
0	90	180	3,730	roll
0	90	0	0,168	pitch
0	-90	0	0,350	roll

Table 14-2: Degree of freedom of joints of robotic arm.

Joint	Lower limit [deg]	Upper limit [deg]	Zero [deg]	Stowed [deg]	Approach Init [deg]
#1	-158	158	0,000	0	0
#2	-158	158	0,000	-90	-45
#3	-158	158	0,000	0	0
#4	-188	128	0,000	-180	-50
#5	-158	158	0,000	0	0
#6	-128	188	0,000	180	-90
#7	-158	158	0,000	0	0

14.3. Options and Trades

With the given design of the robotic arm, a docked capsule is not serviceable. With an arm designed for platform servicing, this task is not feasible due to the required total length and configuration. One option would be to go for an additional Canadarm-sized manipulator near the docking port and integrate a payload handover between the two manipulators.

Table 14-3: Robotic arm trade linear rail vs. cross rail vs. radial rail.

	Linear rail	Cross rail	Radial rail
Design	Easy	Difficult	Easy, like linear rail
Reachability	Good reachability for three faces of the cube Large experiments can block the arm reachability	Very good reachability for three faces Large experiments can block the arm reachability	Good reachability for all faces of the cube Large experiments can be circumvented

14.4. Mass and Power Budget

This section gives an overview on the required elements of this subsystem and lists their masses in order to obtain a total mass budget.

14.4.1. List of Equipment

Table 14-4: Mass budget of the robotics.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
ESP_Manipulator	117.00	10.00	11.70	128.70	42.38
ESP_Manipulator_Gripper	10.00	10.00	1.00	11.00	3.62
ESP_Manipulator_Harness	5.00	10.00	0.50	5.50	1.81
ESP_Manipulator_Rail_Harness	50.00	20.00	10.00	60.00	19.76
ESP_Manipulator_Rail_Mechanism	50.00	20.00	10.00	60.00	19.76
ESP_RCU	10.00	10.00	1.00	11.00	3.62
PL_Manipulator_Control_Station	25.00	10.00	2.50	27.50	9.05

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	267.00			303.70	

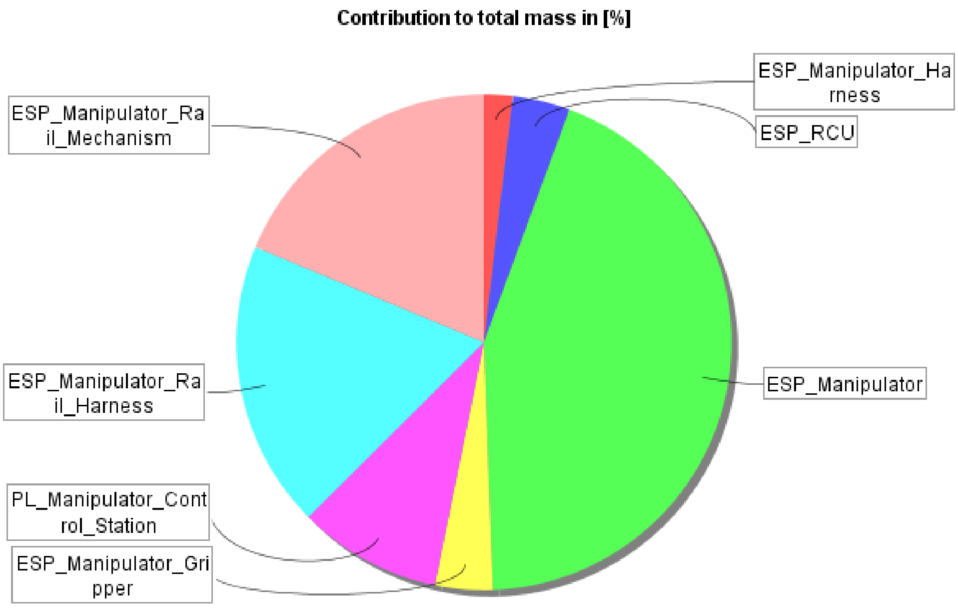


Figure 14-5: Pie chart of mass distribution of the robotics.

14.4.2. Power Budget

This section gives a brief overview of the power consumption of the robotic arm.

Table 14-5: Power budget of the robotic arm.

System Component	Parameter	Unit	Standard Mode	Survival Mode
Robotic_Automation	power_avg_wMargin	Watt	60.200	0.000
ESP_Manipulator	power_avg_wMargin	Watt	16.500	0.000
ESP_Manipulator_Gripper	power_avg_wMargin	Watt	2.200	0.000
ESP_Manipulator_Harness	power_avg_wMargin	Watt	0.000	0.000
ESP_RCU	power_avg_wMargin	Watt	22.000	0.000
PL_Manipulator_Control_Stati	power_avg_wMargin	Watt	16.500	0.000
ESP_Manipulator_Rail_Mecha	power_avg_wMargin	Watt	3.000	0.000
ESP_Manipulator_Rail_Harnes	power_avg_wMargin	Watt	0.000	0.000
Robotic_Automation	power_energy_wMargin	Joule	4369075200....	0.000

14.4.3. Mode dependencies

The arm will only be active during nominal operations. Upon survival mode, the complete subsystem is switched off after reaching a safe manipulator configuration. Subsequently the breaks will hold this position without any power required.

14.5. To Be Further Studied / Additional Considerations

The following point should be further studied in order to obtain a more complete picture of the capabilities and required design of the manipulator:

- Detailed mechanisms to fix experiments and how the robot needs to interact
- Required gripper for this task
- Detailed launch configuration
- Detailed reachability analysis of all possible experiment positions on the platform

14.6. Summary / Comparison to ISS

In summary, the chosen arm design for servicing the experiment platform is much shorter (half the size), much lighter and cheaper (~100 Nm vs. ~1000 Nm) when compared with the Canadarm that is currently mounted on the ISS. In addition, the control mode is more precise and allows environment-compliant robotic operations.

Given the current station configuration and the arm's main task, i.e. servicing the experiment platform, the docked Dragon (or other) capsule is out of reach! One option would be to add an additional Canadarm-sized manipulator at base station for gaining the possibility of adding larger payload (larger than IBDM inner diameter) when already in orbit.

15. AOCS

15.1. Requirements and Assumptions

- On-orbit mass is 25 t
- Constant orbit (400 km, 51.6 deg)
- Free-Flyer is the leading spacecraft; base station is the trailing spacecraft
- Standard orientation is nadir for P/L → drag torque has to be countered

<u>AOCS:</u>

- Based on Base Station AOCS design
- Attitude control in standard mode by CMGs; in survival mode by chemical thrusters
- Attitude determination in standard mode by GNSS; in survival mode by star tracker
- Sensors are distributed among pressurised laboratory and service module
- Actuators are distributed between external science platform and service module
- Not all sensor usable in docked configuration due to obstruction of FOV
- System at least two failure tolerant

15.2. Baseline Design

- External science platform (ESP):
  - CMGs (x2)
  - Rate gyro assembly (x2)
- Service module (SE):
  - CMGs (x2)
  - Sun sensor (x3)
  - Horizon sensor (x2)
  - GNSS receiver (x2)
  - GNSS antenna (x2)
  - Star tracker (x2)
  - (Thrusters for attitude and orbit control on rear end)

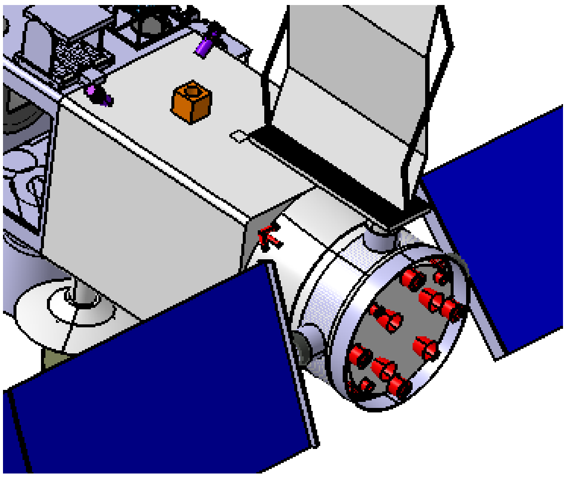


Figure 15-1: AOCS equipment on the Service Module

- Pressurised laboratory (PL):
    - Sun sensors (x3)
    - Horizon sensor (x2)
    - GNSS receiver (x2)
    - GNSS antenna (x2)
    - (Thrusters for attitude control on front end)
    - LiDAR RVD sensor (x2)
    - RVD IMU (x2)
    - Visual camera + lighting (x2)
    - Visual target (x2)

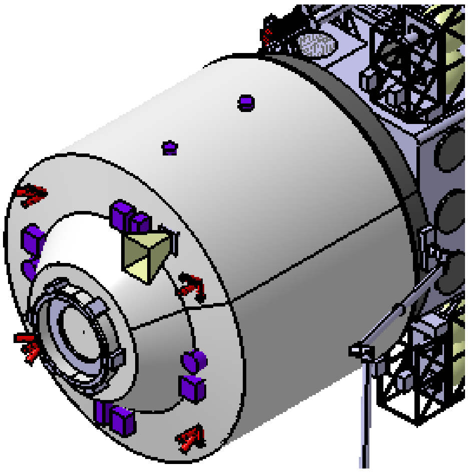


Figure 15-2: AOCS equipment on the Pressurised Lab

15.3. Options and Trades

- Attitude control using thrusters is discarded
    - Impulsive manoeuvres
    - Higher accelerations on payloads during µg phases

15.4. Mass and Power Budget

15.4.1. List of Equipment

Table 15-1: Mass budget of the AOCS equipment.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
ESP_CMG	332.18	20.00	66.44	398.62	45.62
ESP_Rate_Gyro_Assembly	8.40	5.00	0.42	8.82	1.01
PL_GNSS_Receiver	6.00	10.00	0.60	6.60	0.76
PL_GNSS_Receiver_Antenna	1.47	20.00	0.29	1.76	0.20
PL_Horizon_Sensor	7.00	5.00	0.35	7.35	0.84
PL_Rendezvous_Docking_Sensor	16.00	20.00	3.20	19.20	2.20
PL_Rendezvous_IMU	1.50	5.00	0.07	1.57	0.18
PL_Rendezvous_Visual_Camera	0.51	20.00	0.10	0.61	0.07
PL_Rendezvous_Visual_Camera_Light	1.00	20.00	0.20	1.20	0.14
PL_Rendezvous_Visual_Target	2.00	20.00	0.40	2.40	0.27
PL_Sun_Sensor	0.90	5.00	0.05	0.95	0.11
SE_CMG	332.18	20.00	66.44	398.62	45.62
SE_GNSS_Receiver	6.00	10.00	0.60	6.60	0.76
SE_GNSS_Receiver_Antenna	1.47	20.00	0.29	1.76	0.20
SE_Horizon_Sensor	7.00	5.00	0.35	7.35	0.84
SE_Star_Tracker	9.00	5.00	0.45	9.45	1.08
SE_Sun_Sensor	0.90	5.00	0.05	0.95	0.11

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	733.51			873.81	

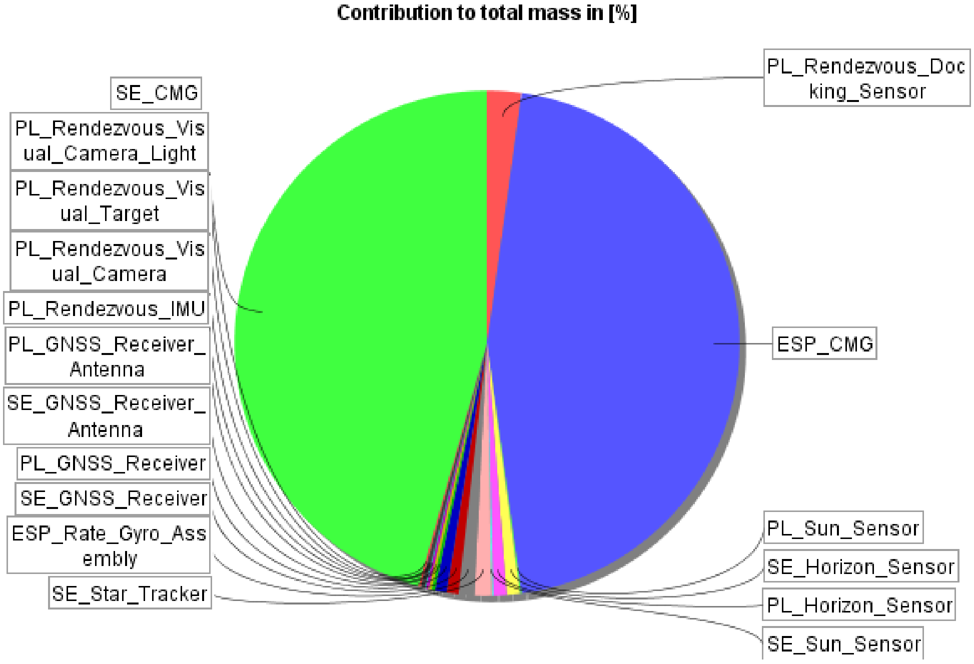


Figure 15-3: Pie chart of mass distribution of the AOCS equipment.

15.4.2. Power Budget

Table 15-2: Power budget of the AOCS equipment.

System Component	Parameter	Unit	Standard Mode	Survival Mode
AOCS	power_avg_wMargin	Watt	423.880	21.000
ESP_CMG	power_avg_wMargin	Watt	163.200	0.000
ESP_Rate_Gyro_Assembly	power_avg_wMargin	Watt	12.600	0.000
PL_GNSS_Receiver	power_avg_wMargin	Watt	16.500	0.000
PL_GNSS_Receiver_Antenna	power_avg_wMargin	Watt	1.200	0.000
PL_Horizon_Sensor	power_avg_wMargin	Watt	0.039	0.000
PL_Rendezvous_Docking_Sen:	power_avg_wMargin	Watt	0.043	0.000
PL_Rendezvous_IMU	power_avg_wMargin	Watt	25.200	0.000
PL_Sun_Sensor	power_avg_wMargin	Watt	0.525	0.000
SE_GNSS_Receiver	power_avg_wMargin	Watt	16.500	0.000
SE_GNSS_Receiver_Antenna	power_avg_wMargin	Watt	1.200	0.000
SE_Horizon_Sensor	power_avg_wMargin	Watt	0.039	0.000
SE_Star_Tracker	power_avg_wMargin	Watt	21.000	21.000
SE_Sun_Sensor	power_avg_wMargin	Watt	0.525	0.000
PL_Rendezvous_Visual_Camer	power_avg_wMargin	Watt	2.101	0.000
PL_Rendezvous_Visual_Target	power_avg_wMargin	Watt	0.000	0.000
PL_Rendezvous_Visual_Camer	power_avg_wMargin	Watt	0.007	0.000
SE_CMG	power_avg_wMargin	Watt	163.200	0.000
AOCS	power_energy_wMargin	Joule	30763492381...	3628800.000

15.4.3. Mode dependencies

For the Survival Mode the power consumption is drastically cut down to only knowledge of orientation, i.e. only the star trackers are on. Re-orientation or attitude stabilisation is done only by thrusters, cp. section 16.4.4.

15.5. Re-Supply Items

- No re-supply items required
- Replacement parts in case of contingency only

15.6. To Be Further Studied / Additional Considerations

- CMGs need to be accurately sized based on spacecraft mass and dimensions
- Simulation of CMG control and desaturation

15.7. Summary / Comparison to ISS

- ISS does not keep constant orbit (orbit raising by visiting vehicle)
- Free-Flyer has smaller and lighter CMGs (currently factor 0.6 including maturity margin) due to station mass
- Sensor number not scalable to station mass
- ISS does not have RVD capability as the active part
- EVA on Free-Flyer only foreseen for contingency

16. Propulsion

16.1. Requirements and Assumptions

- Propulsion for the Free-Flyer needs to support orbit maintenance & transfer, docking, and attitude control during safe mode.
- Propulsion for the Hub will be based on that of the Free Flyer, but detailed in a follow-on study.
- Thrust level and delta-V requirements are provided by AOCS.
- Dimensioning requirement for the bi-prop system is 55 docking cycles.
- Electric Propulsion is used for large manoeuvres (orbit maintenance and transfer including debris avoidance), but inactive in safe mode.
- Propulsion system is one-failure tolerant for all modes and two-failure tolerant against critical or catastrophic events during proximity operations.
- Lifetime is 15 years on station.
- Two or three refuelling missions are acceptable during the lifetime.

16.2. Baseline Design

- The baseline propulsion is a hybrid electric/chemical bipropellant system based on technology currently available or in development.
- In order to reduce launch mass of the Free-Flyer, the chemical propulsion system is designed to carry propellant (for five years, requiring two refuelling missions during a 15-year lifetime → changed after post-processing recalculation of close proximity manoeuvres: higher propellant consumption!). This option was selected during the study to limit mass growth of the overall system.
- Life-limited items have enough copies to ensure redundancy.

Figure 16-1: Propulsion equipment overview (bipropellant system).

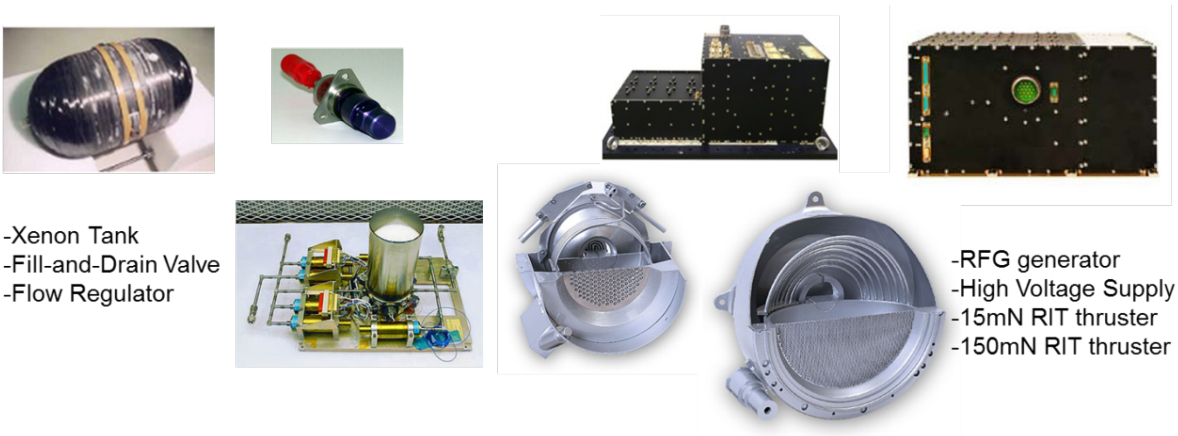


Figure 16-2: Propulsion equipment overview (ion propulsion system).

16.3. Options and Trades

- First trade was between fully-electric, hybrid, or fully bipropellant system; the baseline is now a hybrid system.
- Second trade is between ATV-like full two-failure tolerance and Dragon- or Cygnus-like partial two-failure tolerance; baseline is Dragon approach.
- Third trade was for size and number of propellant tanks: To carry propellant for full 15-year lifetime or rather for 5 or 7.5 years. Baseline is 5 years' worth of propellant.
- Fourth trade was between various tank configurations; baseline is one each fuel and oxidizer tank.

16.4. Mass and Power Budget

16.4.1. List of Equipment and Dry Mass Budget

Table 16-1: Mass budget of the propulsion equipment.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
ESP_Tubing	0.42	20.00	0.08	0.50	0.13
PL_RCS_Thrusters	22.80	5.00	1.14	23.94	6.04
PL_Tubing	1.20	20.00	0.24	1.44	0.36
SE_FDV	0.97	5.00	0.05	1.02	0.26
SE_Flow_Limiters	0.20	20.00	0.04	0.24	0.06
SE_He_Tank_300bar_68l	15.60	5.00	0.78	16.38	4.13
SE_Ion_Prop_Ancillary_HW	10.00	20.00	2.00	12.00	3.03
SE_Latch_Valve_Biprop	5.10	20.00	1.02	6.12	1.54
SE_Main_Thrusters_400N	17.20	5.00	0.86	18.06	4.56
SE_NonReturnValves	0.72	10.00	0.07	0.79	0.20
SE_Pressure_regulator	7.60	10.00	0.76	8.36	2.11
SE_Pressure_Transducer	2.52	20.00	0.50	3.02	0.76
SE_Prop_Filters	0.70	20.00	0.14	0.84	0.21
SE_Prop_Tank_OST25_3	45.40	5.00	2.27	47.67	12.03
SE_Pyrovalves	1.20	10.00	0.12	1.32	0.33
SE_RCS_Thrusters	22.80	50.00	11.40	34.20	8.63
SE_RIT10_EVO	7.20	5.00	0.36	7.56	1.91
SE_RIT10_FCU	1.20	20.00	0.24	1.44	0.36
SE_RIT10_PCDU	6.40	20.00	1.28	7.68	1.94
SE_RIT10_XRFS	14.00	5.00	0.70	14.70	3.71
SE_RIT2X	35.20	20.00	7.04	42.24	10.66
SE_RIT2X_FCU	1.20	20.00	0.24	1.44	0.36
SE_RIT2X_PCDU	36.00	20.00	7.20	43.20	10.90
SE_RIT2X_XRFS	14.00	5.00	0.70	14.70	3.71
SE_Tubing	4.59	20.00	0.92	5.51	1.39
SE_Xe_Tank	78.00	5.00	3.90	81.90	20.67

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	352.23			396.28	

16.4.2. Propellant and Pressuring

- 343 kg Xenon (during post processing here a demand of 656 kg Xenon was estimated for the whole mission lifetime of 15 yrs)
- 757 kg bipropellant during launch (several times refuelling expected)
- 4 kg of pressuring gas

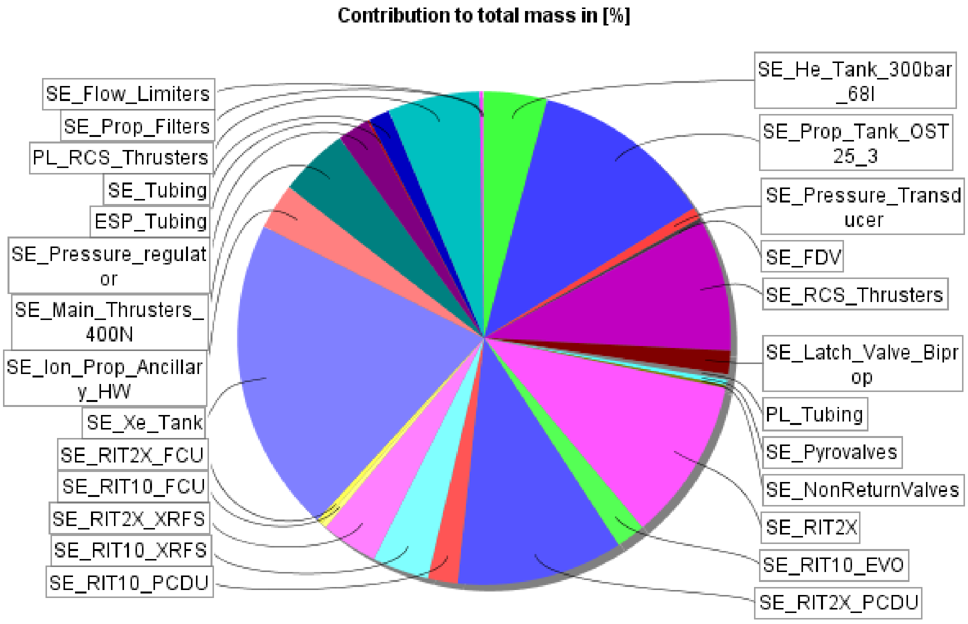


Figure 16-3: Pie chart of mass distribution of the propulsion equipment.

16.4.3. Power Budget

Table 16-2: Power budget of the propulsion equipment.

System Component	Parameter	Unit	Standard Mode	Survival Mode
Propulsion	power_avg_wMargin	Watt	2394.551	3.877
SE_He_Tank_300bar_68l	power_avg_wMargin	Watt	0.000	0.000
SE_Prop_Tank_OST25_3	power_avg_wMargin	Watt	0.000	0.000
SE_Ion_Prop_Ancillary_HW	power_avg_wMargin	Watt	12.000	0.000
SE_Latch_Valve_Biprop	power_avg_wMargin	Watt	0.600	0.600
SE_Main_Thrusters_400N	power_avg_wMargin	Watt	2.738	2.100
SE_Pressure_Transducer	power_avg_wMargin	Watt	0.600	0.600
SE_Pyrovalves	power_avg_wMargin	Watt	0.001	0.001
SE_RCS_Thrusters	power_avg_wMargin	Watt	3.474	0.240
SE_RIT10_EVO	power_avg_wMargin	Watt	166.446	0.000
SE_RIT10_FCU	power_avg_wMargin	Watt	15.120	0.000
SE_RIT10_PCDU	power_avg_wMargin	Watt	21.600	0.000
SE_RIT10_XRFS	power_avg_wMargin	Watt	32.130	0.000
SE_RIT2X_FCU	power_avg_wMargin	Watt	15.120	0.000
SE_RIT2X	power_avg_wMargin	Watt	2031.840	0.000
SE_RIT2X_PCDU	power_avg_wMargin	Watt	58.320	0.000
SE_RIT2X_XRFS	power_avg_wMargin	Watt	32.130	0.000
PL_RCS_Thrusters	power_avg_wMargin	Watt	2.432	0.336
Propulsion	power_energy_wMargin	Joule	17378695514...	669962.880

16.4.4. Mode dependencies

In Standard Mode, the electric propulsion system is active with a duty cycle adapted to the momentary atmospheric drag conditions. The chemical propulsion subsystem is in standby mode and is only active for certain flight phases, in particular docking manoeuvres; a corresponding duty cycle has been budgeted in the average power consumption.

In Survival Mode, the electric propulsion system is off and only the chemical propulsion system is activated as required for the acquisition of Sun-pointing mode or other short manoeuvres.

16.5. Re-Supply Items

- Roughly 30.13 t of bipropellant are expected to be refuelled over the mission life time of 15 years (about 505 kg per rendezvous and docking)

16.6. To Be Further Studied / Additional Considerations

- Architecture of the electric propulsion subsystem
- Redundancy concept should be refined

16.7. Summary / Comparison to ISS

- Far simpler bipropellant system than ATV, due mainly to different implementation of redundancy
- Large delta-V is supplied by electric propulsion
- Electric thrusters allow for establishment of a very good micro-gravity environment

17. Acronyms

Abbreviation	Comments
AOCS	Attitude and Orbit Control System
ATV	Automated Transfer Vehicle
BCDU	Battery Charge Discharge Unit
BS	Base Station
CCU	Command & Control Unit
CE	Concurrent Engineering
CEF	Concurrent Engineering Facility
CFRP	Carbon Fibre Reinforced Polymer
CIGS	Copper-Indium-Gallium-Selenide
CMG	Control Momentum Gyro
CMU	Command & Monitoring Unit
COTS	Commercial Off-The-Shelf
CSA	Canadian Space Agency
CSA-CP	Compound Specific Analyser for Combustion Products
CSS	Chinese Space Station
DAM	Debris Avoidance Manoeuvres
DC	Direct Current
DCSU	Direct Current Switching Unit
DDCU	DC/DC Converter Unit
DHS	Data Handling System
DLR	Deutsches Zentrum für Luft- und Raumfahrt e.V.
DOD	Depth of Discharge
DoF	Degrees of Freedom
EATCS	External Active Thermal Control System
ECLSS	Environmental Control and Life Support System
EDRS	European Data Relay System
EMI	Ernst-Mach Institute (Fraunhofer)
EOL	End of Life

ESA	European Space Agency
ESP	External Science Platform
ESTEC	European Space Research and Technology Centre
EVA	Extra Vehicular Activity (Space Walk)
FF	Free-Flyer
FOV	Field of View
FT	Failure Tolerant
GEO	Geostationary Orbit
GNSS	Global Navigation Satellite System
HST	Hubble Space Telescope
HTV	H-II Transfer Vehicle
IBDM	International Berthing and Docking Adapter
IMU	Inertial Measurement Unit
ISECG	International Space Exploration Coordination Group
ISPR	International Standard Payload Rack
ISS	International Space Station
JAXA	Japan Aerospace Exploration Agency
JEM EF	Japanese Experiment Module External Facility
LAN	Local Area Network
LED	Light-Emitting Diode
LEO	Low Earth Orbit
LEOP	Launch and Early Operations
LOS	Loss of Sight
MBSU	Main Bus Switching Unit
MCG	Momentum Control Gyro
MI-	Mission (Requirements)
MLI	Multi-Layer Insolation
MLT	Mission Life Time
MMOD	Micro-Meteoroid Orbital Debris
MPCV	Multi-Purpose Crew Vehicle
MPPT	Maximum Power Point Tracker
MUMS	Multi-User Facility for High Temperature Material Sciences

NASA	National Aeronautics and Space Administration
OBC	On Board Computer
ORM	Orbit Raising Manoeuvres
P/L	Payload
PBA	Portable Breathing Apparatus (Mask)
PL	Pressurised Laboratory
PPRA	Positive Pressure Relieve Assembly
PV	Photo Voltaic
RAAN	Right Ascension of Ascending Node
REoCV	Reduced End-of-Charge Voltage
RVD	Rendezvous and Docking
RVDS	Rendezvous and Docking System
S/S	Subsystem
SE	Service Module
SLS	Space Launch System
SoC	State of Charge
ST-	Systems (Requirements)
STK	AGI Systems Tool Kit
StM	Standard Mode
SurvM	Survival Mode
TC	Transfer Converter
TCP	Tool Center Point
TCS	Thermal Control System
TDRS	Tracking and Data Relay Satellite
TW	Triebwerk (Thruster)
UTCG	Universal Time Coordinated Gregorian

18. References

[RD 1] L. Di Fino et al. , "Radiation survey in USLab and first measurements in Columbus with the ALTEA detector", Workshop on Radiation Monitoring for the International Space Station, Stillwater, Oklahoma, USA, 2012

[RD 2] L. Di Fino et al. "Measurements on radiation shielding efficacy of Polyethylene and Kevlar in the ISS (Columbus)", Journal of Radiation Research 55, 2014, pp. 64-65

[RD 3] M. R. Shavers et al., „Implementation of ALARA radiation protection on the ISS through polyethylene shielding augmentation of the Service Module Crew Quarters", Advances in Space Research 34, 2004, pp. 1333-1337

[RD 4] ESA ARTES Program (2015), "Ultra Rapid Deployable Antenna (URDA)" [Online] Available: https://artes.esa.int/projects/ultra-rapid-deployable-antenna-urda, [Accessed 07.01.2016]

[RD 5] JAXA Human Space Systems Utilization Mission Directorate, "Kibo Exposed Facility User Handbook" [Online] Available: http://iss.jaxa.jp/kibo/library/fact/data/JFE_HDBK_all_E.pdf, [Accessed 12.01.2016]

[RD 6] Orbital ATK-Goleta, "UltraFlex Solar Array Systems" [Online] Available: https://www.orbitalatk.com/space-systems/space-components/solar-arrays/docs/FS007_15_OA_3862%20UltraFlex.pdf, [Accessed 04.02.2016]

[RD 7] Space Systems Loral, "Award & Launch History – GEO Satellites" [Online] Available: http://www.sslmda.com/html/aboutssl/history_1300.html [Accessed 04.02.2016]

[RD 8] C. Pearson, D. Curzon, P. Blackmore, G. Rao, "Lithium-Ion Small Cell Battery Shorting Study", 20080015786_2008015829.pdf

[RD 9] J. Neubauer, K.L. Ng, A. Benetti, C. Pearson, G. Rao, "Reliability Through Life of Internal Protection Devices in Small-Cell ABSL Batteries", 26 Apr. 2007, 20070032991_2007033024.pdf

[RD 10] J.S. Neubauer, A. Benetti, C. Pearson, N. Simmons, C. Reid, M. Manzo, The Effect of Variable End of Charge Battery Management on Small-Cell Batteries, NASA/TM—2007-215044, AIAA–2007–4789, 20080006656_2008003970.pdf

[RD 11] http://www.boeing.com/assets/pdf/defensespace/space/spacestation/systems/docs/ISS%20Electric%20Power%20System.pdf , [Retrieved 22.01.2016]

[RD 12] Wiley J. Larson, Linda K. Pranke, John Connolly, Robert Giffen, "Human Spaceflight: Mission Analysis and Design", McGraw-Hill Companies; 1 edition (October 26, 1999).

[RD 13] AIRBUS DEFENCE&SPACE, "ELECTRIC PROPULSION THRUSTER FAMILY" http://www.space-propulsion.com/brochures/electric-propulsion/electric-propulsion-thrusters.pdf [retrieved 01 February 2016]

[RD 14] Booth, R. E., & Stoyack, J. E. (1992). Thermal control Coating Selection and Verification for the Space Station Freedom, (1).

[RD 15] BOEING, "Active Thermal Control System (ATCS) Overview" https://www.nasa.gov/pdf/473486main_iss_atcs_overview.pdf [retrieved 01 February 2016]

[RD 16] Fleming, M., Flores, R., & Stoyack, J. (1990). "Design, Verification and Testing of the International Space Station Photovoltaic Radiator", 520–525.

[RD 17] David G. Gilmore, "Spacecraft Thermal Control Handbook, Volume I: Fundamental Technologies" second edition, The Aerospace Corporation, 2002

[RD 18] Krisko, P. H.; NASA's new orbital debris engineering model, ORDEM2010

[RD 19] Christiansen, E.L., METEOROID/DEBRIS SHIELDING, 2003

[RD 20] Christiansen, E.L., et al Space station MMOD shielding, 2009, doi:10.1016/j.actaastro.2008.01.046

[RD 21] Putzar, R., Reducing the vulnerability of space systems, ReVus project, Report E 33/13